EDITORIAL

IGNORANCE + UNBRIDLED ENTHUSIASM = DANGER

THE Jones boy who lives next door is a fine lad, interested in sports, hot rods, outboard hydroplanes, and all of the products of our mechanical, electrical and atomic age. While he is not really one of the meek types, he will grow up to inherit whatever earth we leave him – maybe! He and his friends are now greatly intrigued by our latest endeavours and occasional successes with missiles, rockets, IRBM's, ICBM's, and of course the achievements of Sputniks I and II. The Jones boys of our time feel instinctively that one of those rocket motored missiles could be produced by them very cheaply and easily with a bit of pipe and some fuel compounded from ingredients procured from the gas station and drug store. As they see it, this homemade article can produce a loud noise, a bright flash and, if things really work, can push the bit of pipe up into the air, the higher the better. A number of Jones boys have already entered into the spirit of getting some fun out of this thing.

The newspapers report the incident of two lads who apparently launched a rocket missile that went up to some altitude (the youthful experimenters estimated 8,500 ft – how they estimated is not explained). The experiment was further successful in that the two Jones boys concerned did not blow themselves up nor did the returning pipe damage an innocent citizen or his property. Not so fortunate was the report of another Jones boy's adventure. He, twenty years of age, and reported to be a Laboratory Technician of the National Research Council, suffered severe face and hand burns in a premature explosion of rocket fuel. To quote the newspapers, "He told police he was trying to develop a solid rocket propellent by compressing magnesium in a kitchen meat grinder." The Explosives Division of the Federal Mines Department have announced that youths found experimenting with fuels will be prosecuted under the Explosives Act. This strikes me as a very sensible decision for the protection of all concerned. As far as this aging Jones boy is concerned, the F.M.D. officials can relax, since I am a wholly enthusiastic supporter of the notion that only trained experts should handle such interesting if somewhat unstable compounds, such as concentrated peroxide, fuming nitric acid of any colour, kitchen meat grinders, ground magnesium etc. The Jones boys, with very few exceptions, just do not have enough knowledge of the basic fundamentals to be able to protect themselves against possible dire consequences.

From a dispassionate viewpoint, rocketry experiments by young boys with no scientific training will, without doubt, be of no value to either science or the youthful experimenters. It is a very long odds chance that even a successful and safe pipe rocket launching by a group of boys would so fire ambition that some or any of the group from that point on would dedicate his future toward the necessary scientific training required to qualify him as a chemist, physicist or engineer. A thinking man, in pursuing an objective, accepts the attendant hazards only after weighing the possibilities of failure or success. When there is no well-defined objective, indulging in a potentially dangerous enterprise is at best foolhardy.

To my way of thinking, there is absolutely no point in encouraging young lads to experiment with rockets, even under the guidance of their school teachers. Greater purpose would be served if we could interest them in a desire to further their education along the lines of science and engineering. Above all, we should try to explain to them, in a way they will understand, that backyard experiments of this nature are not only without value to themselves or anyone else but are, in addition, very dangerous to themselves and others. A dead boy is a forever lost asset to the world. A dreadfully maimed boy is pitifully handicapped throughout his life.

E. B. SCHAEFER
Chief Engineer
Canadair Limited

WIND TUNNELLESS TEST

Figure 1

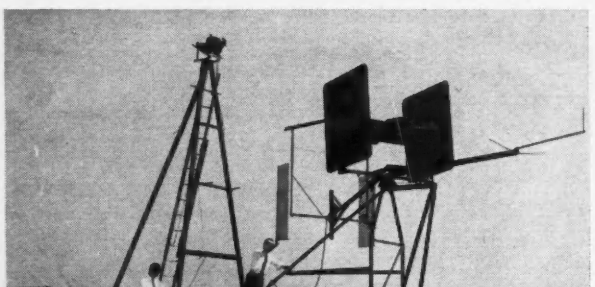

Figure 2

SOME interesting work has been done by the De Havilland Aircraft of Canada Ltd. in connection with open air tests on models. Mr. R. D. Hiscocks has given us the following account.

Truck-mounted models are shown in Figures 1 and 2. At 60 mph airspeed a Reynolds number is obtained which is quite acceptable for low speed studies. Figure 1 shows an 11 ft model of one of the earlier versions of the Caribou and the purpose of this test was to measure downwash in the region of the tail. (Note the wool tufts.) Figure 2 shows an engine nacelle model mounted on a stub wing with end plates. The guide vanes in this case were provided to make the model weathercock into wind. Most of the work on these models comprised flow studies and cameras can be seen mounted at various points of the compass to record the behaviour of the wool tufts. The tuft technique was found very simple, flexible and convenient.

Figures 3 and 4 show 11 ft models of the Caribou mounted on the back of an Otter. The rather redundant-looking supporting structure was provided to facilitate the measurement of various force components. As with the truck-mounted models, the bulk of the tests were made to obtain qualitative rather than quantitative data, but some quite useful numbers were obtained, numbers which agreed quite well with refined data from wind tunnel sources.

A great advantage of these methods is that they provide raw data which is immediately useful to the Design Office without elaborate corrections.

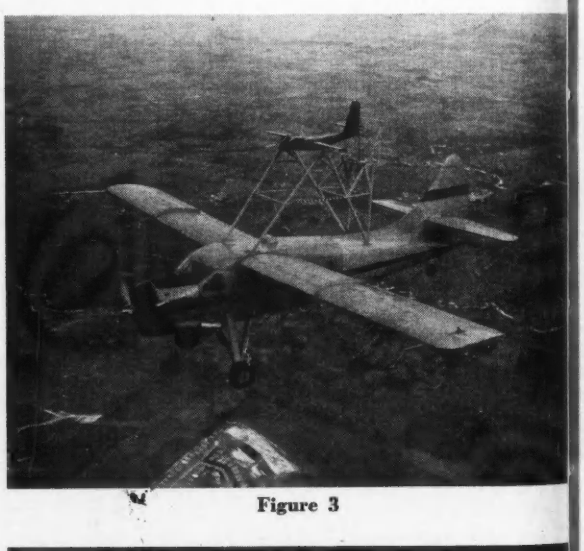

Figure 3

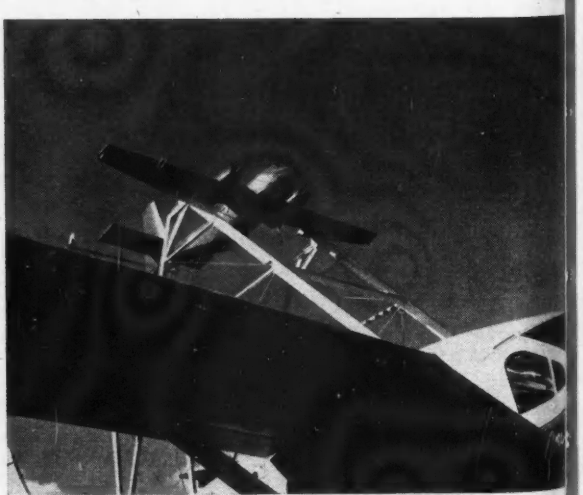

Figure 4

AEROPHYSICAL PROBLEMS OF FLIGHT AT EXTREME ALTITUDES AND SPEEDS

*Continued from Page 10**

Part 2. HYPERSONIC EFFECTS

CHARACTERISTICS OF HYPERSONIC FLOW

The aerophysical problems of flight at extreme speeds arise from two sources: (a) large Mach number and (b) high stagnation temperature. The regime of hypersonic flow is taken to include both of these effects.

The term "hypersonic" was introduced by Tsien in a paper dealing with the similarity rule for flows around bodies at high Mach numbers[26]. This paper initiated a series of investigations based on simplifications of supersonic theory provided by the assumption of a large Mach number. A detailed summary of theoretical developments in hypersonic flow has been prepared by Lees[27].

Tsien showed that the potential flow of a gas at very high Mach numbers over geometrically similar airfoils and bodies is dynamically similar if the product of Mach number and thickness ratio remains constant[26]. In other words the basic characteristic parameter is the ratio of maximum velocity normal to the surface to ambient sound speed. The similarity rule applies at high flight Mach numbers to slender bodies at low angles of attack since the local value of M remains high and the shock and Mach waves are approximately parallel to the direction of flight.

In an attempt to allow for the fact that the shock wave is close to the body but not coincident with the surface, the "tangent-wedge" and "tangent-cone" approximations were developed for hypersonic flow around slender airfoils and bodies, respectively. In the tangent-wedge approximation the pressure on the surface element having an inclination θ_b is considered to be the same as the pressure on a semi-infinite wedge of half-angle θ_b at the same flight Mach number.

The validity of the linearized theory of supersonic flow at high Mach numbers and the accuracy of the hypersonic approximation have been considered by von Kármán[28]. The tendency of the linearized theory to depart from the physical facts at high Mach number is seen by a consideration of the shock wave at the nose of a body. According to linearized theory, the shock wave is a Mach wave having an inclination independent of the flow deflection and dependent only on the Mach number. On the other hand, according to the hypersonic approximation, the angle of the shock wave is proportional to the angle of the surface and tends to become independent of the Mach number. Since the flow around a wedge can be determined exactly, a comparison can be made as indicated in Figure 8. Clearly the linearized theory departs appreciably from the exact theory for a Mach number of the order of 5 or greater while the hypersonic approximation is nearly correct. It appears from these results that hypersonic flow may be said to begin somewhere between $M = 3$ and $M = 5$.

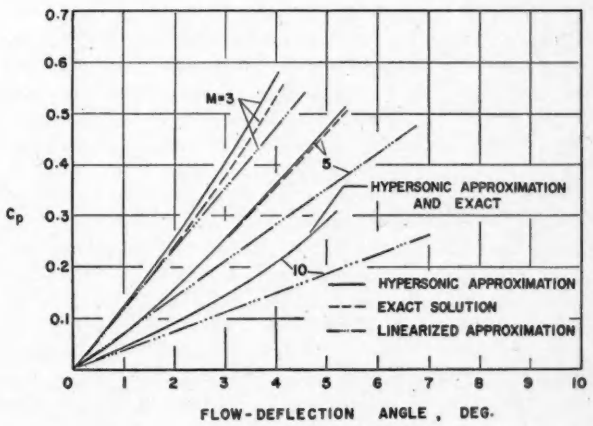

Figure 8
Calculated pressure coefficients for two-dimensional flow around a wedge (Reference 3)

Another hypersonic approximation is based on Newton's particle concept[29, 30]. It is to be expected that the limit of a gas flow as $M \to \infty$ described above would apply at any density. The effect of rarefaction is to hasten the approach to the limiting flow. However, the Newton particle theory differs from the free molecule limit discussed in that, although the reflection process involves the destruction of the component of particle momentum normal to the surface of a body, it assumes that the tangential component of momentum remains unchanged[31]. The reflection is no longer diffuse but has a preferred direction which is the same for all molecules. The coefficient of normal pressure remains the same as that given in Eq. (16). For continuum flow in two

*Canadian Aeronautical Journal, January 1958.

dimensions around a flat plate we have on the upstream-facing side of a flat plate

$$C_p = (\gamma + 1) \sin^2 \theta \qquad (17)$$

where $\gamma \to 1$ as $M \to \infty$ and on the downstream-facing side, $C_p = 0$. As the Mach number increases, the shock wave on the front face lies nearer and nearer the surface, corresponding to increasing pressure, and the expansion wave on the rear face produces a decreasing pressure[32]. The use of Newton's theory as an approximation to hypersonic flow appears to be considerably better for three-dimensional bodies than for two-dimensional shapes.

The characteristics of hypersonic flows are further illustrated by wind tunnel experiments. Tests on wings in hypersonic tunnels show that the flow tends to become two-dimensional at high Mach number because of the large sweep-back of the Mach lines. Even on wings of low aspect ratio, two-dimensional flow occurs over most of the surface, the sidewash being confined to small regions near the wing tips. These facts provide the basis of a theoretical method for calculating hypersonic flow around three-dimensional bodies developed by Eggers and Savin[33]. Measurements of the effect of high Mach number on the pressure distribution around a wing of low aspect ratio show that the lift depends mainly on the increased pressure on the lower surface. The upper surface makes no important contribution since the reduced pressure differs little from the ambient stream pressure. Tests on bodies of revolution indicate that the base pressure is less significant at high Mach numbers. The pressure drag results largely from the elevated pressures over the forepart of the body. The pressure on the sides toward the rear of the body are nearly the same as the ambient pressure and the expansion of the flow around the base produces little change.

The above characteristics may be described as "Mach number effects". In hypersonic flight the kinetic energy of approach of the air relative to the aircraft is of sufficient magnitude to produce real gas effects. The fundamental assumption of the kinetic theory of gases is that the internal energy of the gas is contained in the random translational motion of the molecules. As the temperature of the gas rises above 1,000°K, it becomes increasingly apparent that considerable internal energy is also stored up in the internal motions of the molecule (rotation, vibration), and in the excited electronic states and ionization. At high temperature part of the internal energy of the gas will arise from dissociation and the formation of new components. During a transition in the macroscopic properties of a gas in a flow, the gas requires a finite "relaxation time" to reach equilibrium in both composition and distribution of energy between the various degrees of freedom of its molecules. Below a Mach number of 6 and an altitude of 40,000 ft (12 km), no appreciable relaxation effects occur in flight since $T < 1{,}000°K$ and the internal energy of the gas is contained in the translational and rotational degrees of freedom. This is the regime of supersonic flight. When $T > 1{,}000°K$, relaxation effects are important and we enter the hypersonic region of flight for which $7 < M < 24$ ($2{,}000°K < T < 8{,}000°K$).

PHYSICAL PROPERTIES OF AIR AT HIGH TEMPERATURE

We shall consider this subject in three parts: (a) the distribution of the internal energy of air in equilibrium, (b) properties of undissociated air at high temperature, and (c) properties of air in dissociation equilibrium.

Let us first consider the distribution of the internal energy of air in equilibrium. At high temperatures (above 1,000°K), the internal energy of a gas arises from the translational, rotational and vibrational energies of the molecules, excited electronic states, dissociation, formation of new components and ionization. The rotational degrees of freedom of a molecule are excited at very low temperatures (as low as 10°K). On the other hand, the vibrational degrees of freedom are excited at high temperatures only and dissociation, ionization etc. are high temperature effects.

The distribution of the total equilibrium internal energy among the various components of air is shown in Figure 9 (see Reference 34), which illustrates the conditions behind a normal shock at an altitude of 100,000 ft (0.1 atmos). As the Mach number ahead of the shock wave (or flight Mach number) increases, the percentage of energy in molecular translation and rotation decreases steadily from 90% at $M = 6$ to 30% at $M = 24$. Less than 50% of the internal energy is in the translational and rotational degrees of freedom above a temperature of 5,000°K. It should be noted that the change in composition during dissociation is similar for all pressures, but the temperature range of the variation is smaller for lower pressures. Below 2,500°K the internal energy is distributed among the molecules of nitrogen, oxygen and nitric acid. In the range $3{,}000°K < T < 6{,}500°K$ the internal energy is divided essentially between nitrogen molecules and oxygen atoms. When $T > 8{,}000°K$ the internal energy is contained mainly in the nitrogen atoms.

The calculations of internal energy on which Figure 9 is based have included the energies associated with all degrees of freedom of the molecules and the dissociation energy for N_2, O_2 and NO, but no allowance is made for the excited electronic states of oxygen or nitrogen atoms since this contribution is extremely small. The relative significance of vibration excitation

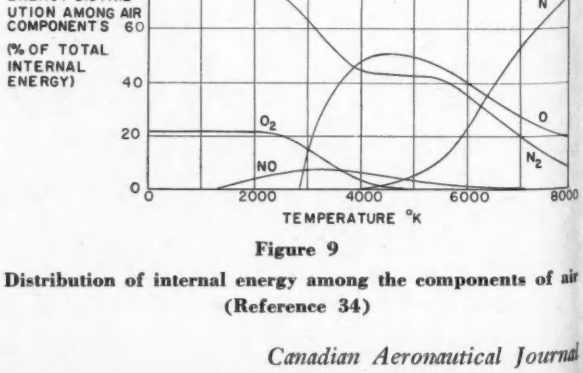

Figure 9

Distribution of internal energy among the components of air (Reference 34)

and dissociation is made apparent by these calculations. For $T < 2{,}000°K$ the nitrogen vibration excitation dominates. In the range $3{,}000°K < T < 6{,}500°K$ the contribution of the dissociation of oxygen to the internal energy is more important and for $T > 6{,}500°K$ the dissociation of nitrogen makes the largest contribution.

Ionization will not be important for temperatures below about 10,000°K, and this effect, therefore, lies outside the temperature range under consideration in this review. The electrostatic forces present in an ionized gas will introduce some large effects. For example, under an inverse square law of interaction between ionized molecules, the viscosity coefficient will vary as the 5/2-power of the temperature. Since an ionized gas is electrically conducting, magneto-electric fields and forces can be induced.

We consider next the properties of undissociated air at high temperature. The effects of high temperature on the equation of state for an undissociated gas can be neglected. Thus $p = \rho RT$ and $dE = C_v\,dT$, where E is the internal energy. These relations hold even when C_v and C_p vary with temperature.

Where viscosity and diffusion are concerned, the internal degrees of freedom of the molecule are not significant since only the transport of momentum and mass are involved. In fact, deductions from the kinetic theory still apply. In the simple kinetic theory a molecular collision is represented by the interaction of two smooth elastic spheres and it is found that $\mu \sim \sqrt{T}$. This variation is not accurate at ordinary temperatures but it is expected that it will apply at very high temperatures. If the molecules are assumed to be rigid spheres surrounded by a weak force of attraction (Sutherland's model[35]), then $\mu \sim T^{3/2}/(T + \text{constant})$. This relation for the dependence of viscosity on temperature is extensively used and should be satisfactory for undissociated air from room temperature up to about 2,000°K. A more complex model consisting of a point center of attractive force at short range (Lennard-Jones' model) has been made the basis of extensive calculations of viscosity[36, 37]. It is possible to calculate with reasonable accuracy the momentum transport properties of gases and gas mixtures with any number of components. Such calculations are limited by the lack of data regarding molecular structure at very high temperatures. Empirical force constants are limited to those derived for relatively low temperatures.

Measurements of viscosity at high temperature have been made by Vasilesco[38]. His results indicate a close agreement between these experiments and the theory based on Sutherland's model.

The diffusion of one component of air through another becomes important when relaxation times are widely different or in oxidation or burning problems. Only a few determinations have been made of the coefficient of diffusion of nitrogen-oxygen mixtures and this information is insufficient to establish a reliable molecular model. Some estimates of the coefficient of diffusion have been made in Reference 39 based on viscosity data and the Lennard-Jones model. For all practical purposes, the coefficient of diffusion is independent of the properties of the components of the mixture.

In aerodynamic flows a high temperature is frequently accompanied by large gradients which will produce a significant diffusion effect. Again, further information relevant to molecular force constants is needed for reliable estimates.

At high temperatures the vibrational degree of freedom of the molecule will make a contribution to the specific heat of a gas. If it is assumed that the vibrational motion of the molecule can be adequately represented by a simple harmonic oscillator, then for a gas in vibrational equilibrium[40]

$$\left.\begin{aligned} C_v &= C_{v0}\left[1 + (\gamma_0 - 1)\frac{a^2e^a}{(e^a - 1)^2}\right] \\ C_p &= C_{p0}\left[1 + \left(\frac{\gamma_0 - 1}{\gamma_0}\right)\frac{a^2e^a}{(e^a - 1)^2}\right] \\ \gamma &= 1 + \frac{1}{\left(\frac{1}{\gamma_0 - 1}\right) + \frac{a^2e^a}{(e^a - 1)^2}} \end{aligned}\right\} \qquad (18)$$

where $a = \Theta/T$, Θ = molecular vibration temperature, and the subscript 0 specifies that the value of the relevant quantity is that for a calorically perfect gas.

The effects of electronic energy levels, distortion, and anharmonicality on the specific heat have been considered[40] and found to be small for temperatures up to 5,000°K. For $T < 5{,}000°K$, the estimates of specific heat which allow only for the vibrational degree of freedom correspond quite well with available data. At temperatures higher than 5,000°K, the above effects could become appreciable. Accurate estimates must await more detailed information about the possible molecular energy levels at high temperatures.

Little is known about the thermal conductivities of gases at temperatures above room temperature since they are very difficult to measure. The theory is also complicated by the fact that the internal degrees of freedom of the molecule affect the energy transport process. The interchange of energy in molecular collisions involving the internal degrees of freedom is a slow process at ordinary temperatures and relaxation effects are important. The cases of very slow and rapid transfer have been adequately considered[39]. The latter case appears to be approximately true for diatomic molecules at high temperature.

For temperatures above 1,000°K the Prandtl number ($Pr = \mu C_p/\kappa$) attains a maximum value of 0.73 and then tends asymptotically to a value of 0.71. Owing to uncertainties in the extrapolation of μ, C_p, and κ to very high temperatures, it is acceptable to take a constant value for Pr above a temperature of 1,500°K.

The physical properties of air extrapolated to 12,000°K are tabulated in Reference 41. This data can

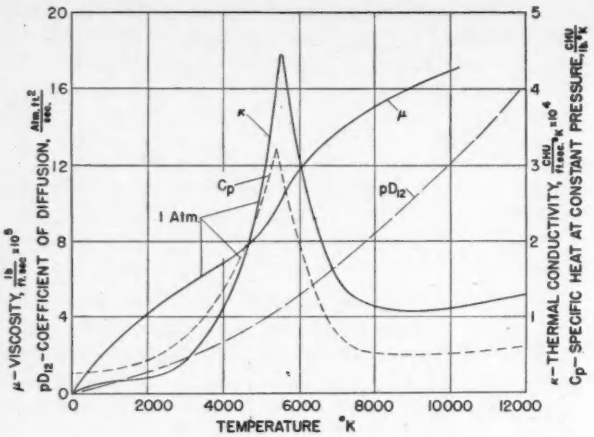

Figure 10

Properties of air in dissociation equilibrium (Reference 41)

be represented adequately between 2,000°K and 8,000°K by the following relations,

$$\left.\begin{aligned} \mu &= 6.03 \times 10^{-5}\ (T/4000)^{0.527} && \text{lb/ft sec} \\ C_p &= 0.3046\ (T/4000)^{0.021} && \text{CHU/lb °K} \\ \kappa &= 2.569 \times 10^{-5}\ (T/4000)^{0.548} && \text{CHU/ft sec °K} \end{aligned}\right\} \quad (19)$$

Finally we consider the properties of air in dissociation equilibrium. The effect of dissociation on the momentum transport process arises from the changes it produces in the local gas composition. The coefficient of viscosity can be determined by considering the transport properties of a mixture of six components. However, the force constants for the atomic species are not known well enough to justify the exact treatment. It is sufficient for an acceptable approximation to assume that the three atomic components belong to one group and the three molecular components to a second group. Assuming smooth molecules and force constants interpolated from data given in Reference 42, we obtain the viscosity curve shown in Figure 10 (see Reference 41). The molecular weight was taken as the mean value for the respective group.

The diffusion of the atomic group through the molecular group can be determined and the curve shown in Figure 10 produced[41].

Approximate values for the thermal conductivity have been obtained assuming a constant value of the Prandtl number (0.715) and very rapid transfer. Approximate determinations of the heat capacities for translational and internal degrees of freedom deduced from the data of References 40 and 43 indicated that the Prandtl number for dissociated air varied between 0.6 and 0.9[41]. A typical curve for the variation of κ with T for a pressure of 1 atmos is shown in Figure 10. The probable variation of the specific heat of air in dissociation equilibrium is also indicated[42, 43]. The variation of the speed of sound in air in dissociation equilibrium with the temperature can now be determined[41].

RELAXATION EFFECTS

In hypersonic flow changes of the state of the gas are exceedingly rapid, and the time involved can become comparable with the time required for a gas to reach equilibrium in both composition and distribution of energy between the various degrees of freedom of its molecules, achieved by the mechanism of molecular collisions. At room temperature the transfer of translational and rotational energies is readily accommodated since only two or three collisions are required. On the other hand many thousands of collisions are needed to adjust the vibrational degrees of freedom and the transfer of vibrational energy is very difficult. Also, only a few molecular encounters involve sufficient energy to produce dissociation and relatively rare three-body collisions are required to produce the recombination of dissociated atoms.

The relaxation time (or time lag in adjustment) is the time required for the adjustment of each degree of freedom to $(1 - 1/e)$ of the final equilibrium value. Relaxation time data are presented in Figure 11.

Vibrational relaxation times have been determined using the spark interferometer technique to examine the density behind plane shock waves moving into oxygen and nitrogen, and schlieren techniques to determine the thickness of the shock wave in oxygen[44, 45, 46]. Both measurements agree with the theory of Landau and Teller that $\tau \sim 1/\sqrt[3]{T}$ where τ is the relaxation time. As indicated in Figure 11, the relaxation time for O_2 vibration lies between 10^{-5} and 10^{-7} seconds if the temperature is between 1,000°K and 3,000°K.

A second relaxation zone was observed when the temperature was in the neighbourhood of 3,500-4,000°K[45] and it was concluded that this related to the dissociation of oxygen. For the temperature indicated, the relaxation time was about 10^{-6} seconds. Problems relevant to the theoretical relaxation times calculated for oxygen dissociation are discussed in Reference 34. This reference presents a theory consistent with experimental measurements and uses it to estimate the relaxation time for the dissociation of nitrogen (Figure 11).

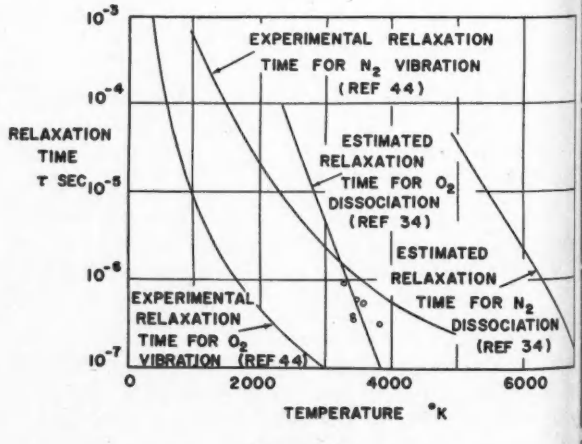
EXPERIMENTAL RELAXATION TIME FOR N2 VIBRATION (REF 44)
ESTIMATED RELAXATION TIME FOR O2 DISSOCIATION (REF 34)
ESTIMATED RELAXATION TIME FOR N2 DISSOCIATION (REF 34)
EXPERIMENTAL RELAXATION TIME FOR O2 VIBRATION (REF 44)
RELAXATION TIME τ SEC
TEMPERATURE °K

Figure 11

Relaxation times for air referred to the density at one atmosphere (Reference 34)

When the relaxation time is small, the effects of this phenomena can be accounted for by introducing a bulk viscosity. This quantity is also called volume or compression viscosity and it is assumed to vary with temperature in the same way as the ordinary shear viscosity. Perhaps the major effect of the bulk viscosity is to alter the equation of state of the gas to the following form[6],

$$p = \rho RT - \mu_b \frac{\partial u_k}{\partial x_k} \qquad (20)$$

In other words bulk viscosity effects occur in flows in which the dilatation $\left(\frac{\partial u_k}{\partial x_k}\right)$ is large such as the normal shock wave. According to Reference 47, the ratio of bulk viscosity to the ordinary shear viscosity (μ_b/μ) can be related to the relaxation time associated with the internal degrees of freedom of the molecule provided the energy exchange involving these degrees of freedom is rapid (easily accomplished by molecular collisions), that is, if the relaxation time is small. Continuum considerations of bulk viscosity give $\mu_b/\mu = \frac{2}{3}$ and it is found in Reference 47 that this implies that about 5 molecular encounters are required to reduce a sudden disturbance in the rotational degrees of freedom of air molecules to $1/e$ of its initial value.

THE SHOCK WAVE IN HYPERSONIC FLOW

The effect of high initial Mach numbers on the properties of a normal shock wave at various altitudes is illustrated in Figure 12. This figure shows the variation of air temperature behind a normal shock wave with the initial (flight) Mach number for a range of ambient pressures (altitudes) at a fixed ambient temperature of 222°K[3]. The states of the gas behind the shock wave are assumed to be in equilibrium. The upper curve indicates the temperature attainable behind the shock front if the air retained the properties of a perfect gas, that is, a gas in which the internal energy remains essentially in the active degrees of freedom (translation and rotation). The lower curves show for a range of ambient pressures (or altitudes) the temperatures which would result in a real diatomic gas in which energy has been absorbed in exciting vibration, dissociation, and

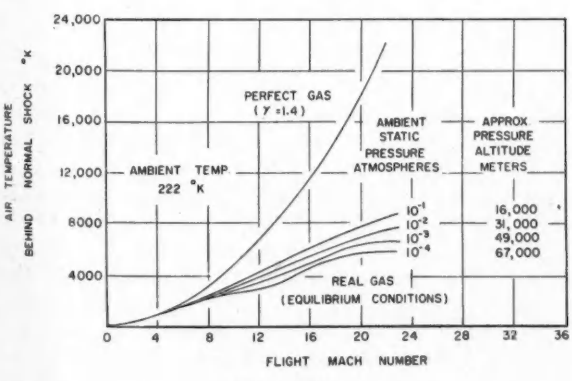

Figure 12

Air temperatures behind a plane shock wave in a real diatomic gas (Reference 3)

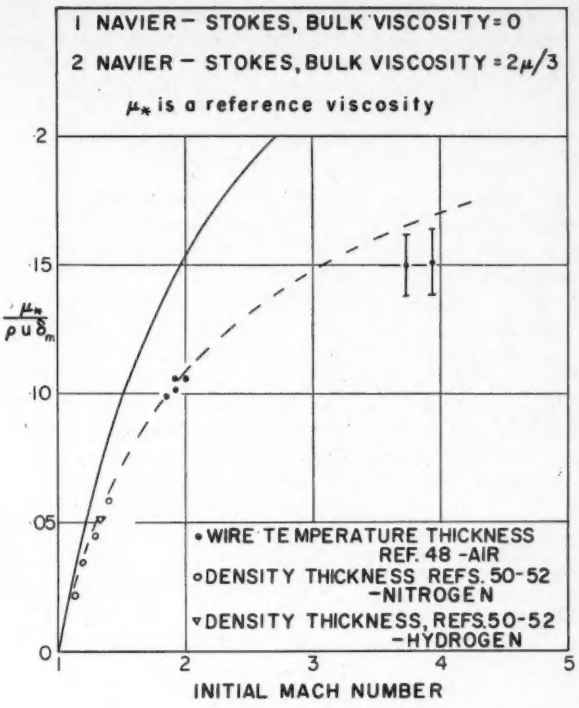

Figure 13

Variation of shock thickness with Mach number (Reference 48)

ionization. Figure 12 suggests that real gas effects begin to appear above a Mach number of about 5. It will be noted that the effect of higher altitude is to hasten the onset of real gas effects.

Investigations of the internal nature of the shock wave have been made in the lower range of Mach numbers ($1.1 < M < 4.5$). The transition in temperature through the normal shock wave obtained by an equilibrium temperature probe has been determined by Sherman[48]. The experimentally determined transition was compared with the variation of temperature through the shock wave calculated on the basis of the Navier-Stokes equations with and without bulk viscosity (for further details and references see References 6 and 49). From profiles of this type the thickness of the shock transition was found and a similar comparison between theory and experiment was made (Figure 13). The thickness is based on the maximum slope of the transition curve and it is plotted in reciprocal form against the Mach number upstream from the shock front. Shock thicknesses obtained by Greene, Cowan and Hornig[50, 51, 52] are also shown in Figure 13. These investigators used the measurement of the optical reflectivity of shock fronts in a shock tube to determine the shock thickness.

It is shown in Figure 13 that the Navier-Stokes equations are sufficient for the description of the transition through the shock front in air for upstream Mach numbers up to 2. The equations must include a bulk viscosity, $\mu_b = \frac{2}{3}\mu$, to account for the effects of rotational relaxation phenomena which tend to increase the thickness of the wave.

When the upstream Mach number is very large, the initial directed (macroscopic) energy of the air is converted into energy associated with the random translation, rotation, vibration, dissociation, electronic excitation and ionization of the molecules. When the shock transition is such that vibration and dissociation play an important role, the wide difference between the relaxation times of these phenomena and the translational and rotational relaxation times becomes apparent. The translational and rotational (active) degrees of freedom of the molecule may be considered to adjust instantaneously to equilibrium because of their relatively short relaxation times. Therefore the temperature through and just behind the shock zone is the ideal gas value. Subsequently there is a decrease in the temperature as the (inert) degrees of freedom associated with vibration and dissociation reach equilibrium in the order given. In the range of Mach numbers from 6 to 20 and at an altitude of 120,000 ft, the relaxation path lengths for the inert degrees of freedom in the uniform region downstream from the normal shock wave to reach full thermodynamic equilibrium are less than 1 ft. Small fractions of dissociation (order of 1%) involve relaxation path lengths over 5 ft. At lower altitudes the higher density will produce shorter relaxation path lengths for dissociation[34].

HYPERSONIC EXPANSION FLOWS

Let us consider the effect of relaxation in a Prandtl-Meyer expansion fan. The time of transit through the expansion wave can be very short (but finite) just outside the boundary layer on a body and the transit time becomes longer as the distance from the body increases. Relaxation effects will therefore be concentrated in the region at the base of the fan. At an altitude of 120,000 ft with an initial velocity of 10,000 ft/sec and initial temperatures ranging between 1,000°K and 4,000°K, the relaxation path lengths for the inert degrees of freedom can vary between 1/10 ft and 100 ft. Under these conditions the transit time for a fluid particle 1 ft from the body is as low as 10^{-4} seconds and we conclude that the zone of relaxation effects can extend some distance from the body[34].

As in the case of the normal shock wave the transit time near the body can be so short that only the active degrees of freedom are adjusted to equilibrium throughout the flow and the expansion takes place isentropically with the participation of the active degrees of freedom only. This is called the frozen flow condition. The temperature in the flow field is determined by the temperature of the active degrees of freedom. The frozen energy will not be in equilibrium with the isentropic temperature and at some later time, depending on the relaxation time for the inert exchanges, the frozen energy will be transmitted to the gas at the new temperature by molecular encounters. This process of adjustment involves an irreversible addition of heat to the flow accompanied by an increase in entropy.

The analysis of adiabatic expansions with chemical reactions taking place suggests that the quantity $\frac{1}{T}\frac{dT}{dt}$ is a reciprocal characteristic time for the flow process[53]. When reactions are sufficiently rapid that near-equilibrium flow conditions occur, the temperature lag (ΔT) is given approximately by

$$\frac{\Delta T}{T} = \left(\frac{1}{T}\frac{dT}{dt}\right) t_{\max} \tag{21}$$

where T is the actual value of the local temperature, and $t_{\max}$ is the maximum value of the reaction time for the chemical process.

This criterion can be applied to flow fields, such as the Prantdl-Meyer flow, to determine the regions in which conditions of near-equilibrium flow are obtained. It is considered that if

$$\left(\frac{1}{T}\frac{dT}{dt}\right) t_{\max} \leq 10^{-3} \tag{22}$$

then equilibrium flow can be maintained, that is, the relaxation time for the inert degrees of freedom is very much less than the relevant transit time[34].

A calculation of a Prandtl-Meyer flow for a deflection of about 20° indicates that large decreases in pressure and temperature will occur as the result of a frozen flow condition ($M_1 \doteq 4$, $T_1 \doteq 4{,}000°K$). It is of interest to investigate thin airfoils and blunt bodies with a view to a qualitative assessment of the importance of relaxation effects on aircraft design. No large relaxation effects are expected in the flow around thin airfoils at small angles of attack. For example if $\theta = 0$, $M_1 = 20$, the temperature behind the oblique shock wave at the leading edge of a 10° double wedge airfoil is less than 1,100°K. At this temperature the only inert degree of freedom likely to be affected is the vibration, and the associated energy will be small. The expected increase in drag due to the lag in the adjustment of the vibrational degree of freedom through the Prandtl-Meyer fan is less than 1%.

On the other hand thin airfoils at large angles of attack are subject to large losses of lift due to frozen flow through the Prandtl-Meyer fans. For example, if $\theta = 5°$, $M_1 = 20$, a 5° double wedge airfoil can have a 10% loss of lift. These results are based on the assumption that real gas equilibrium is achieved behind the oblique shock wave which precedes the expansion fan. This assumption may not be realized since the relaxation path length for the inert degrees of freedom behind a normal shock wave is greater than 10 ft at $T_1 = 1{,}000°K$ at an altitude of 120,000 ft.

In the case of blunt bodies the high temperatures behind the detached bow wave suggest that full thermodynamic equilibrium can be attained in the distance of detachment. This is the case for a dissociation greater than 2% at altitudes of 120,000 ft. On a blunt body a very rapid expansion of the flow occurs behind the bow wave. Departures from local equilibrium flow over blunt bodies have been estimated in Reference 34. These departures were based on the value of the product of the relaxation time and the reciprocal characteristic flow time (Eq. (22)) and evaluated for the flow around a sphere. Under hypersonic conditions equilibrium flow for all the inert degrees of freedom can be maintained to about the 40° position around the circumference from the

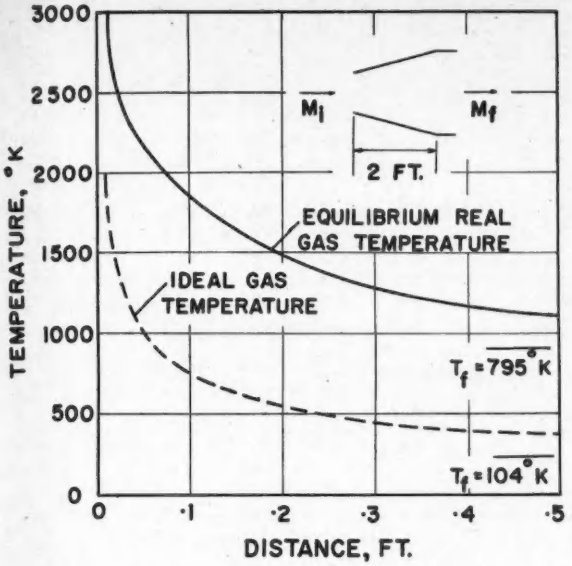

Figure 14
Temperature variation in a hypersonic nozzle (equilibrium flow conditions: $M_i = 2.46$, $M_f = 8$, $p_i = 0.229$ atm, Reference 34)

stagnation point. At the 60° position frozen flow has developed, the product $\frac{\tau}{T}\left(\frac{dT}{dt}\right)$ being of the order of 1. The frozen flow does not radically alter the distribution of the Reynolds number per foot or the local Mach number around the circumference. However, the isentropic gas temperature can be about 20% lower than the real gas equilibrium temperature. The simulation of the flows associated with hypersonic flight in a wind tunnel requires the reproduction of both the Mach number and the stagnation temperature of flight. The expansion flow in the nozzle can be accompanied by very high cooling rates such that the criterion given in Eq. (22) is by no means fulfilled and frozen flow must occur.

The hypersonic flow in a nozzle 2 ft long has been considered in Reference 34. An inlet temperature of 3,825°K and inlet and test section Mach numbers of 2.46 and 8, respectively, were assumed. A nozzle with two plane diverging walls and two parallel walls was considered using a one-dimensional theory. The estimated isentropic and equilibrium real gas temperature distributions along the nozzle are indicated in Figure 14. The isentropic drop in temperature is considerably more than the real gas value. A consideration of the relaxation time and the transit times indicates that no equilibrium flow can be expected near the nozzle inlet.

If it is assumed that the early flow in the nozzle is frozen (where very rapid expansions occur) and that adjustment begins in the low temperature regions just before reaching the test section, an estimate can be made of the effect of the large departure from equilibrium flow. It is found that the Mach number in the test section ($M = 8$ for isentropic flow) can be reduced by as much as 50% by the relaxation effect which raises the temperature above the ideal gas value. Also the entropy is increased as a result of the irreversible addition of heat. From the design point of view it is evident that nonequilibrium processes should be avoided if possible both in the wind tunnel and in flight.

THE BOUNDARY LAYER IN HYPERSONIC FLOW — AERODYNAMIC HEATING

Rocket engines now under development will open the way to flight with hypersonic range up to a Mach number of 25. The full potential of such powerplants will be realized only if the problem of aerodynamic heating can be satisfactorily solved for each aircraft design. The importance of aerodynamic heating is clearly seen in the fact that it exercises control over the configuration of the aircraft, the available flight corridor and the type of construction[54]. Local heating rates must be understood by the aerodynamicist so that he can shape the aircraft to minimize their effects. They must also be appreciated by the structural designer who must allow for the accompanying thermal stresses.

Let us consider the heat balance for a small portion of the aircraft skin. We shall assume steady state conditions. Then the rate of convective heat transfer into the surface due to aerodynamic heating is equal to the rate of dissipation of heat outward from the surface by radiation plus the rate at which heat is carried away by a cooling system i.e.

$$q_c = q_r + q_i \tag{23}$$

where we assume that the section of skin is so thin that we may neglect the heat conduction into and out of it.

If no internal cooling is provided ($q_i = 0$) then the convective heat transfer into the surface is balanced by the radiant heat transfer away from the wall. The resultant skin temperature for this case is called the radiation equilibrium temperature (T_r). If neither internal cooling nor radiation occur, the skin attains a temperature termed the equilibrium or recovery temperature (T_e). Then for heat balance, neglecting internal cooling ($q_c = q_r$), we have

$$\rho V C_p (T_e - T_w) St = \epsilon B T_w^4 \tag{24}$$

where

$$St = H/\rho V C_p \tag{25}$$

is called the Stanton number. Now the recovery temperature is given by

$$T_e = T_1\left[1 + r\left(\frac{\gamma - 1}{2}\right) M_1^2\right] \tag{26}$$

and it is apparent that for hypersonic flow T_e will be a large quantity. The emissivity (ϵ) is limited by Kirchoff's law to a value of 1 or less and for aircraft materials ϵ may be taken to be about 0.8. It will be seen that the radiation equilibrium temperature can be minimized only by maintaining the lowest possible values for air density and the Stanton number. Density effects have been considered above. It is clear that one of the basic boundary layer problems is the study of the dependence of the Stanton number on variables over which the designer has control.

The Stanton number will be influenced by a large number of parameters. Such quantities as Mach number,

Reynolds number and Prandtl number will be important parameters. Other effects will arise from gradients of density, pressure and temperature and from the ratio of surface temperature to the temperature at the edge of the boundary layer (T_w/T_1). The shape of an aircraft must be such that no local areas of high density occur (Eq. (24)). It is evident that the structure of the boundary layer as it moves back along the aircraft depends on its history, which is affected by the pressure and temperature gradients, and again body shape becomes a critical factor. Further details about the above effects in laminar boundary layers may be found in References 55, 56, 57 and 58.

One of the most important factors upon which the Stanton number depends is the Reynolds number at which the laminar boundary layer changes to a turbulent boundary layer. The variation of Stanton number with the Reynolds number is shown in Figure 15 (see Reference 54). The change from laminar to turbulent flow in the boundary layer can produce a 5-fold increase in convective heat transfer. The primed quantities on Figure 15 refer to the reference temperature method developed by Rubesin and Johnson[59] in which the reference temperature at which the properties of the air are defined is expressed as a function of local Mach number and T_w/T_1[54].

The determination of the transition Reynolds number is still a difficult problem. The Tollmien-Schlichting theory of the stability of the boundary layer has been extended by Lees and Lin[60] and Van Driest[61]. Although no exact relation exists between boundary layer instability and the transition Reynolds number, the two phenomena can be related qualitatively if transition occurs essentially as the result of small flow disturbances. However, in practice, transition from laminar to turbulent flow in the boundary layer is more likely to be controlled by surface roughness, external disturbances

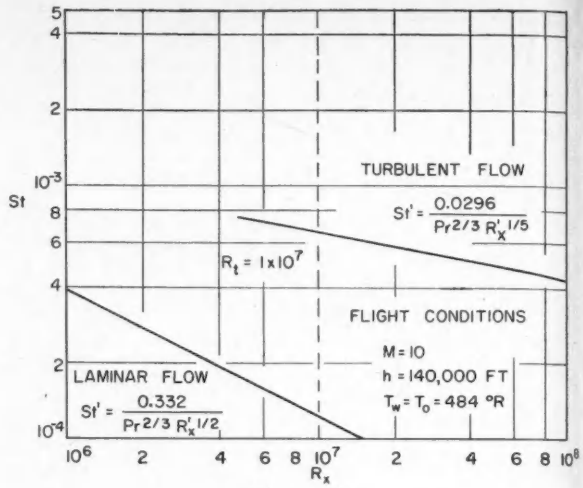

Figure 15

Variation of the Stanton number with Reynolds number for a flat plate (Reference 54)

due to noise, shock impingement etc. To the designer the choice of a transition Reynolds number is essentially a guess even though the difference between laminar and turbulent heat transfer rates has a considerable effect on the trajectory and maximum allowable speed.

The boundary layer is also subject to real gas effects. The high temperatures developed in the boundary layer can result in dissociation and ionization of the air. A reference enthalpy method has been developed by Romig and Dore[62] which takes account of local Mach number and wall-to-free-stream temperature ratio. The effect of dissociation is to reduce the heat transfer rate. However, further research is needed in order to calculate heat transfer rates at high temperature with confidence.

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(To be continued in the next issue)

AMENDMENT OF THE BYLAWS

The amendments to the Bylaws, voted upon last November and carried, as reported on page 32 of the January issue of the Journal, have now received the approval of the Secretary of State. The Council has approved corresponding changes in the Regulations.

The amendments cover the change of name of the grade of "Technician" to become "Junior Member" and a general revision of the scale of membership dues. Both will become effective with the beginning of the new fiscal year, on the 1st April, 1958.

Copies of the Bylaws and Regulations embodying these amendments are available from C.A.I. Headquarters and from Branch Secretaries, on request.

GLOBAL NAVIGATION IN HIGH SPEED AIRCRAFT†

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Air transport navigation has undergone few changes since the years immediately following World War II. The opening up of polar routes has introduced gyro steering and a change in direction referencing; however most airmen consider these methods to be peculiar to high latitudes only and with no general application. There is a possibility that some day these techniques, modified and made automatic, may form the basis for navigating high speed aircraft in all latitudes. The gradual increase in the speed of transport aircraft has had no appreciable effect on navigation techniques either. Within a few years, however, speeds of 400-600 kts will be reached which coupled with long range flights will bring about changes in the navigation equipment and techniques used on global routes. Emphasis will be placed on self-contained aids for en route navigation and because of a requirement for greater accuracy and stability the greatest change will probably be in heading systems.

SELF-CONTAINED AIDS

The global air route pattern, which has remained more or less static for a number of years, has recently undergone changes with the introduction of long range aircraft. Refuelling stops, which were important a few years ago, are being by-passed in favour of more direct and economical routes. New routes over remote areas are also appearing. As traffic increases and more routes are added to the already complex pattern, there will be a greater demand for a single global facility for en route navigation which will provide area coverage for all latitudes rather than designed for any particular route pattern. This is a practical requirement and undoubtedly, some day, a ground-based global navigation aid of this type will appear. In fact, there are a number of aids under development, such as Dectra, Delrac, Consolan, Radio Web, Navaglobe and Navarho, which may meet this requirement. However, as no one particular aid appears to stand out above the others, there is no general agreement on the facility to be adopted. As it is unlikely that a general agreement will be reached until high speed, long range aircraft have been in operation for some years, we are faced with the problem of navigating aircraft the design and speed of which have outstripped the development of an en route global navigation and traffic control facility.

Today, ground based facilities for en route navigation are found only along the older and more congested air routes and even then large overwater sections are inadequately covered for precise navigation when only these aids are used. Until a world-wide facility is introduced, high speed aircraft flying global routes will require self-contained aids for efficient en route navigation to support the operational flexibility required to fly the most economical path between terminals. This is not practicable if aircraft are equipped to use only facilities peculiar to any one area or flight path. Fortunately for the commercial operator, the military have an even greater requirement for flexibility in air transport operations. Consequently, a considerable effort has been made to develop navigation aids which are independent of ground based facilities.

Since the introduction of pressure pattern techniques near the end of World War II, the three most important developments in the field of aids for en route navigation have been:

(a) utilizing the doppler principle for navigation,
(b) perfecting the periscopic sextant and the introduction of new astronomical tables, and
(c) development of reliable lightweight search radar.

Until a common system is available, doppler navigation, search radar, astro, and an improved heading system which will be discussed later, will be the main en route navigation aids for high speed aircraft flying global routes.

Doppler Navigation

The practical application of the doppler principle to navigation is probably the greatest single post-war development in self-contained aids. Basically a doppler navigation system is composed of a microwave transmitter-receiver, antenna and associated computer. The total weight of the transport version will probably be far less than 100 lb. Ground speed and drift angle information, which can be displayed on dials, is measured continuously either over sea or land. Thus, such information combined with true air speed and heading provides a continuous indication of the aircraft's position in either latitude and longitude, or polar co-ordinates.

†Paper read at the Joint I.A.S./C.A.I. Meeting in Montreal on the 21st October, 1957.

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depending on the type of computer preferred. Furthermore the wind velocity can also be displayed. For the first time in the history of air navigation, up-to-date wind information is continuously available under all weather conditions. Hence, better use can be made of jet streams and they also can be avoided more easily.

Doppler navigation being a precise form of dead reckoning, errors are accumulative and unless corrected with other information the accuracy of the indicated positions deteriorates as the flight progresses. Ground speed can be measured to 1% and drift angle to at least 0.5° and, as a result, errors will not become excessive until a considerable distance has been flown. Consequently, doppler positions over the mid-Atlantic should be consistently more accurate than those available from existing facilities. Accurate heading information is very important because it is one of the components used by doppler to compute positions. Magnetic headings, at the present time, will not provide the accuracy required to fully exploit the accuracy capabilities of doppler, particularly in high speed aircraft. This problem will be discussed later.

Since doppler is a vital component of the navigation system required for flying global routes, there is no doubt that civil versions of the military equipment will be available when high speed aircraft come into service. In fact, in response to demands from aircraft manufacturers and operators of future long range, high speed aircraft, some aspects of doppler navigation developments were declassified by the armed services in June 1957.

Airborne Search Radar

Lightweight search radar, which is becoming increasingly more common in transport aircraft, will perform two roles in the new high speed transports. These are:

(a) *A supplement to doppler navigation.* Over areas such as found on arctic routes, radar fixes can be obtained at frequent intervals, in which case a combination of doppler navigation and search radar will result in very precise en route navigation. Although search radar is of little value over the ocean, it will provide landfall fixes for resetting the doppler dead reckoning computer.

(b) *Cloud Warning.* Severe weather conditions are associated with vertical-type clouds which, in some parts of the world, will often extend above the altitudes flown by jet transports. Radar will aid in avoiding these clouds. This is a very important function of search radar.

Astronomical Observations

Today, with navigation aids dominated by electronic systems, one is liable to get the impression that astronomical observations no longer have a place in air navigation. This is not so since astro is used to bridge the gaps on routes where electronic aids are not available, while in other areas it is a standby facility. Another important role played by astro is to check the compass heading. Indeed, even with the improved compass systems available today, the importance of this role has not diminished since in high speed aircraft a heading error of several degrees will result in a track error of many miles in a short time.

The periscopic sextant, that came into use several years after the war, is a decided improvement over the older bubble sextants. It can also be used for checking the heading, thus eliminating the requirement for an astro compass which has been a standard item for many years in transport aircraft flying global routes.

An advance in sight reduction tables has been made within recent years; in 1953 the new Sight Reduction Tables for Air Navigation (AP 3270 in the US H0249) were introduced. These tables, designed specifically for use in the air, eliminate several steps previously required in astro solutions. Three volumes cover all latitudes.

The requirement for astro to check heading and act as a standby aid will not be lessened when doppler navigation supplemented by search radar is used in high speed, long range aircraft. Although the periscopic sextant and the new sight reduction tables are adequate, some changes in astro techniques will be necessary to keep pace with the higher speeds. Time consuming three-star fixes will be replaced by observing only the celestial body required to check either the track-made-good or the ground speed, whichever is needed. In many cases the procedure will be to compute astro solutions in advance of the observation time.

In any event, astro will be a much more reliable aid than in the past since nearly all of the en route flying in high speed aircraft will be above the layer-type clouds.

HEADING SYSTEM

Let us turn our attention to the heading system required for high speed aircraft flying global routes.

In general terms, in all modern compass systems the directional reference for the autopilot is provided by a gyro slaved to the local magnetic meridian by a pendulously mounted flux valve. Several compass systems, such as those used on arctic routes, incorporate a switch to "cut out" magnetic slaving, so that the free gyro can be used to maintain direction. Although magnetic compass systems are reliable and appear to provide an accurate and stable heading reference, this is not the case on closer examination.

Limitations of Magnetic Reference

Variation

Magnetic information used in air navigation is prepared from surface data and errors varying from one or two degrees to many times these values are frequently encountered at flight altitude in many regions of the world. Furthermore, magnetic storms increase the magnitude of errors in high magnetic latitudes.

The normal practice on long range flights has been to check the compass by astro every hour and after large heading alterations to ensure that the proper heading has been flown. Although some airmen may not have realized it, the distance flown during the hour set the requirement for a heading check every hour. In the past this distance was not sufficient to produce large track errors if the heading were checked hourly. How-

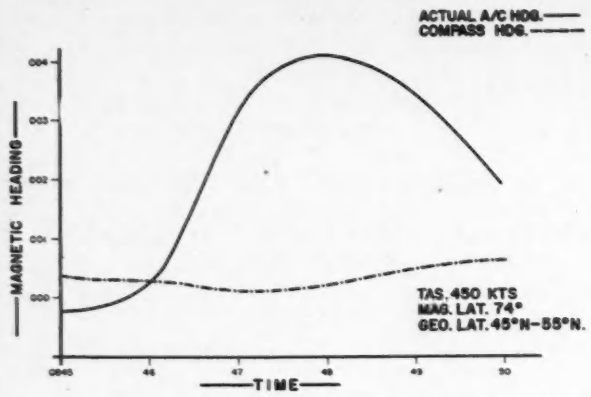

Figure 1

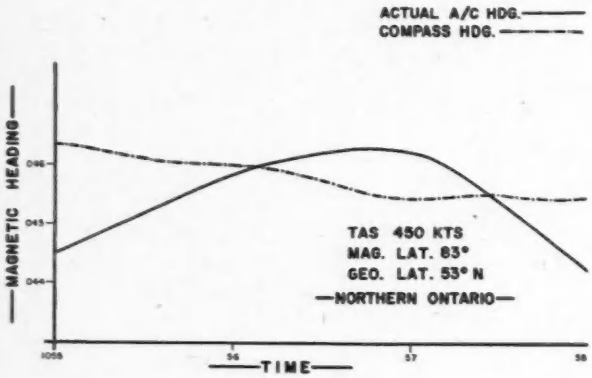

Figure 2

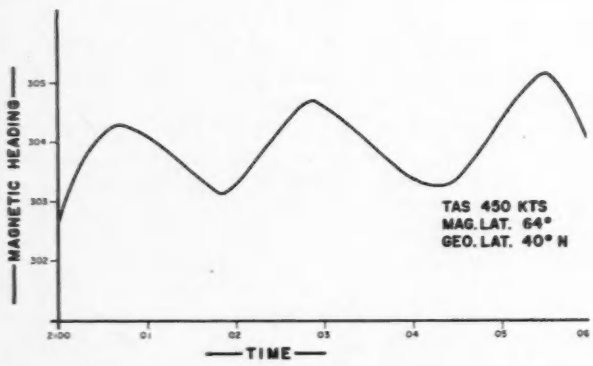

Figure 3

ever, flying at 540 kts a check every 20 min is required if the same heading accuracy is to be maintained as when flying at 180 kts with a check every hour. This is an appreciable increase in work load.

The rate of change of variation is another factor to be considered when using a magnetic reference. For example, when flying between Montreal and Goose Bay in present-day aircraft the rate of change of variation is about 5° per hour but in high speed transports of a few years hence it will be 15° per hour, a change in heading of 5° every 20 min. It is obvious that automatic variation correction is required if a magnetic reference is used in high speed aircraft.

Northerly Turning Error

When using magnetic reference, heading oscillations on northerly flights and altitude hunting and changes in air speed on east-west flights will cant the pendulous flux valve about a N-S axis. When this occurs, long period heading wander is induced in the flux valve by the vertical component of the earth's magnetic field. The period and amplitude of the oscillation increase with an increase in speed and magnetic latitude.

This type of directional instability, a form of northerly turning error, has been reported by commercial airlines on flights from Scotland to Iceland. The RCAF investigated this problem in 1951 using a North Star aircraft. Oscillations of 4-5 min duration with amplitudes of 10° were recorded between Ottawa and Churchill. More recently, the USAF has observed the same effect. Figures 1, 2 and 3 show the degree of heading instability that can be encountered over Canada and the North Atlantic when flying at approximately 450 kts. Astro observations used as position lines to check track are also seriously affected by heading oscillations, as shown in Table 1.

It is obvious that magnetic compass systems will not provide the heading accuracy required to efficiently use doppler navigation or even astro as a standby aid in high speed aircraft. The inherent limitations already mentioned indicate that no further development of magnetic referencing systems is warranted. Therefore, a major change in the concept of heading systems is required to provide accurate and stable directional reference.

Gyro Reference

As a result of technical advances in recent years, directional gyros in use today have real wander rates of not more than 1° to 2° per hour. These gyros are the only reliable directional instrument available when crossing the Arctic where the magnetic compass cannot

TABLE 1
ASTRO OBSERVATION ERROR

Ground Speed Kts	Heading wander — degrees per minute									
	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
400	2′	4′	6′	8′	11′	13′	15′	17′	19′	21′
425	2	4	7	9	12	14	16	18	20	23
450	2	5	7	9	12	14	17	19	21	24
475	2	5	7	10	12	15	17	20	22	25
500	3	5	8	11	13	16	18	21	24	26
525	3	6	8	11	14	17	19	22	25	28
550	3	6	9	12	14	17	20	22	26	29
575	3	6	9	12	15	18	21	24	27	30
600	3	6	9	13	16	19	22	25	28	32

be used. The slaving of these gyros to the magnetic field in almost any latitude deteriorates their performance and results in a less accurate and stable heading than if they were "free" and checked once an hour by astro.

The heading system required for high speed aircraft flying global routes should consist of a directional gyro with a real wander rate of 1° or less per hour, provision for latitude compensation and automatic correction for meridian convergency. With this system the flux valve would be used only as a standby during the en route phase of the flight and if so desired could be switched into the system to provide magnetic heading in terminal control areas and for taking off. One astro heading check per hour would be sufficient regardless of geographical position or aircraft speed once the gyro had been properly aligned to the heading.

CONCLUSION

Until a common aid is available high speed aircraft flying global routes will require self-contained aids for efficient navigation. Doppler radar will be the primary aid and its introduction will be a major advancement in air transport navigation.

However, probably the greatest advancement in navigation in many years will occur in heading systems. The gyro will become the primary directional instrument and the earth's magnetic field will be a secondary aid. The basic concept of directional systems and steering techniques that will be used eventually in all latitudes by high speed aircraft will have originated in northern latitudes where through necessity the gyro has been the only practical directional aid regardless of aircraft speed.

McCURDY AWARD

The McCurdy Award will be presented at the Annual General Meeting on the 26th - 27th May, 1958.

It is presented each year

To A Resident of Canada,

For Achievement in design, manufacture or maintenance related to aeronautics.

NOMINATIONS ARE INVITED

Each nomination should include

(a) The name and affiliation of the nominee,

(b) Confirmation that he is a resident of Canada,

(c) A citation of the particular achievement for which the nomination is being put forward, and

(d) The name of the nominator.

The nominee need not be a member of the C.A.I. and the achievement need not have taken place within the last year, though it should be recent.

Nominations should be in the hands of the Secretary not later than the 15th March, on which date they will be handed over to the McCurdy Award Selection Committee.

STABILITY AND CONTROL CHARACTERISTICS OF THE VERTICAL ATTITUDE VTOL AIRCRAFT†

by E. R. Hinz*

Ryan Aeronautical Company

SUMMARY

The concept of the vertical attitude VTOL aircraft has now become a fully established fact through successful flights of manned research aircraft. One of the major problem areas satisfactorily solved in this development was that of stability and control at low speeds and in hovering. Pertinent aerodynamic characteristics at angles of attack above stall are presented in this paper, and their influence on flight in the transition regime is discussed. Maneuvering control criteria are analyzed to indicate requirements for the low speed regime, and it is shown how a reaction control system may be combined with aerodynamic controls for low speed flight. Thrust control characteristics peculiar to hovering and transition flight are illustrated. Use of stability augmentation is discussed and a comparison made with the requirements of contemporary high performance aircraft. The dynamic flying characteristics in transition are shown through the use of time history studies of take-off and landing maneuvers.

INTRODUCTION

In this age when we have begun to talk seriously about earth satellites and lunar travel, it would seem highly improbable that one could generate technical or popular interest in low speed flight. But such has not been the case since the introduction of the USAF Model X-13 vertical take-off and landing airplane. Built and flown by Ryan under U.S. Air Force sponsorship, this research airplane has evoked great interest in both the technical and popular fields as leading the way to a new freedom of flight.

The X-13 airplane was designed and built solely for the purpose of establishing the feasibility and developing the techniques of vertical take-offs and landings and transitions to and from conventional flight. Although high speed flight was not an objective, it was always considered that the resulting configuration should be compatible with projected design philosophies of future aircraft which need VTOL capabilities. With these ideas in mind, the X-13 evolved into the configuration shown in Figure 1.

This airframe has a 60° delta high wing incorporating integral fuel tanks. Because of the short tail moment arm, the vertical tail is necessarily large and, in addition, wing-tip fins are provided to improve directional stability. Control in the conventional flight regime is provided by trailing edge elevons and a conventional rudder. Control in transition and hovering is attained through use of reaction jets. Since this design was to have maximum compatibility with high speed tactical aircraft design, it was decided to use the primary propulsion (and lifting) jet for pitch and yaw control, thereby making the most efficient and compact design possible. Auxiliary wing-tip jets, using the relatively cool bleed air from the compressor as a working medium, provide roll control.

Figure 1
USAF X-13 airplane

Thrust for this airplane is provided by a Rolls-Royce Avon engine with sufficient thrust to permit VTOL operations at altitudes above sea level and at moderate ambient temperatures. The overall thrust-to-weight ratio provides sufficient thrust to demonstrate the level of maneuverability in transition and hovering which is required of a tactical VTOL airplane.

†Paper read at the Joint I.A.S./C.A.I. Meeting in Montreal on the 22nd October, 1957.

*Unit Chief, Systems and Controls.

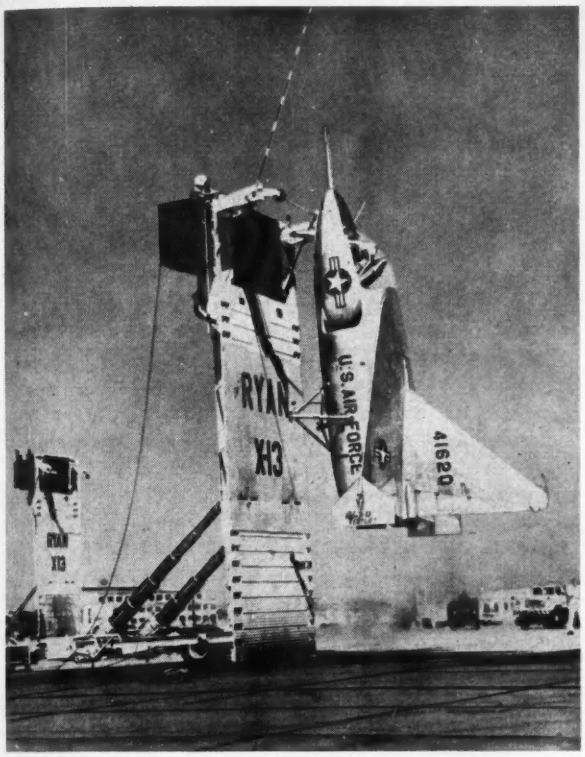

Figure 2
X-13 prior to vertical take-off

Since a prime objective of VTOL airplane design is to eliminate the unnecessary airborne weight not required for flight, essentially all of the landing and take-off provisions for the X-13 remain on the ground in a mobile service unit. The only provisions retained in the airframe for take-off and landing are the nose hook and landing struts. (With minimum drag not a prime design objective for this aircraft, no attempt was made to retract or otherwise streamline the simple landing gear.) The X-13 in take-off configuration is shown just prior to ascent in Figure 2. A complete description of the development of the X-13 is contained in Reference 1.

The X-13 airplane has more than fulfilled its objective of demonstrating the feasibility of hovering and transition flight. It has shown that the vertical attitude hovering and landing VTOL is a practical configuration whose important features can readily be incorporated into a tactical airplane design. This concept enables the tactical design to be optimized around its required mission performance with negligible compromise for landing considerations.

The purpose of the present paper is to illustrate certain aspects of vertical attitude VTOL airplanes which are unique to this concept. Some of these areas will be documented with pertinent data from Ryan studies.

STABILITY AT HIGH ANGLES OF ATTACK

Of primary consideration in the design of the vertical attitude VTOL is the selection of a suitable wing planform. Recent aeronautical history records some interesting discussions within the industry over the merits of straight, swept, and delta planforms for high speed flight. Although these discussions never resolved the wing selection problem, certain aspects of the delta-wing appeared useful for a vertical attitude VTOL airplane. These can be summarized as follows:

(1) A stable pitching moment up though maximum lift with no abrupt loss of lift after the maximum is reached.
(2) Absence of abrupt unsymmetrical stalling of the wing at high angles.
(3) No horizontal tail to suffer the adverse effects of wing downwash. (This is also one less element to optimize for both high and low speed flight.)
(4) Trailing edge aerodynamic controls on a delta-wing are effective to very large angles of attack.
(5) The delta-wing has obvious structural advantages and provides a usable fuel storage space.

It is also apparent by observation of some contemporary high performance aircraft that the delta-wing has proven suitable for high speed flight. A recent paper by R. L. Lickley and L. P. Twiss of the Fairey Aviation Company[2] documents the adaptability of the delta-wing to high speed flight.

The aerodynamics of the delta-wing are quite well known in the conventional flight regime. However, only recently have extensive aerodynamic studies been conducted to learn more about the configuration at angles of attack above stall. In these studies, the body axes reference system was used in place of the better-known stability axes and is illustrated in Figure 3. Wind tunnel tests of delta-wing VTOL configurations have been made for angles of attack from 0 to 90°. Special tests at $\alpha = 90°$ with the model rolling about its longitudinal (vertical) axes yielded aerodynamic characteristics representative of hovering in measurable winds.

The longitudinal behaviour is obviously of first importance in the study of the vertical attitude VTOL airplane. The lift coefficient trend through the transition region is shown in Figure 4. Maximum trimmed lift is

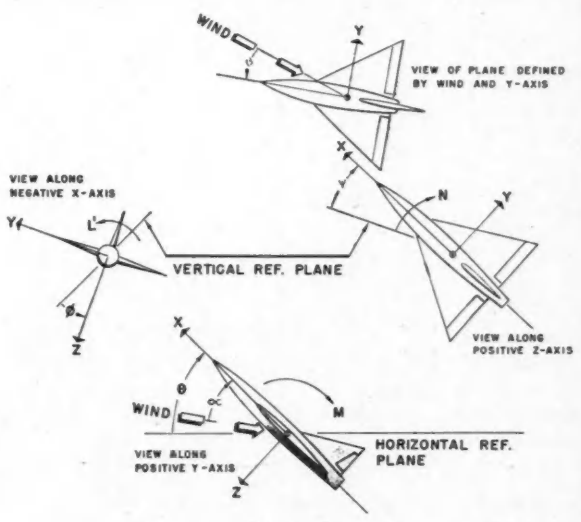

Figure 3
Body axes system

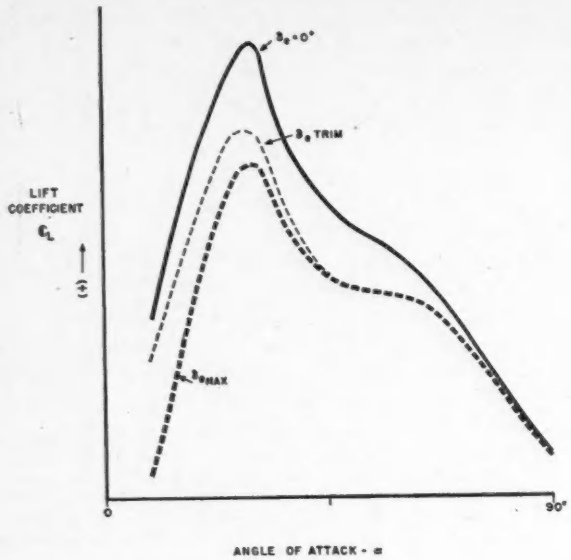

Figure 4
Lift coefficient

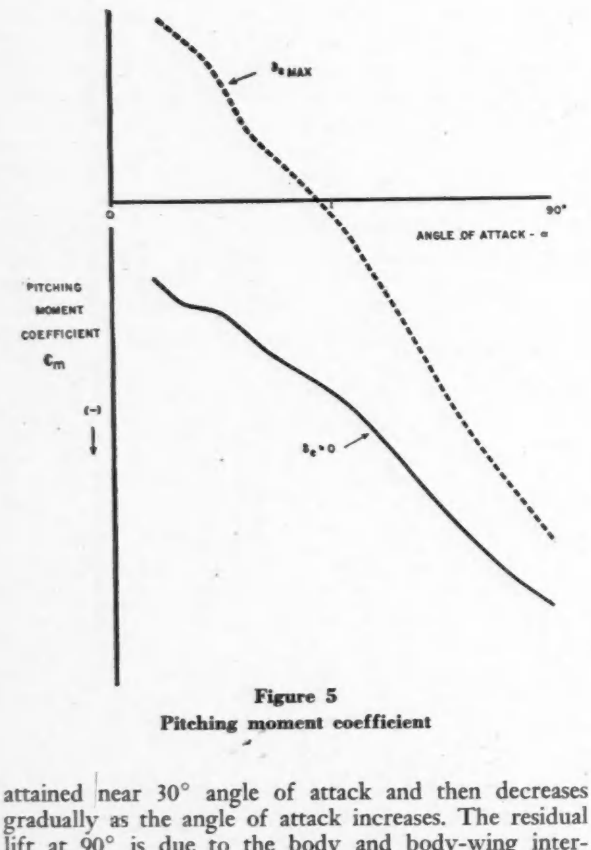

Figure 5
Pitching moment coefficient

attained near 30° angle of attack and then decreases gradually as the angle of attack increases. The residual lift at 90° is due to the body and body-wing interference effects. It is obvious that no abrupt fall-off in lift occurs throughout the transition angle of attack region.

A stable pitching moment exists up to 90° angle of attack with no sudden changes, as indicated in Figure 5. The elevons are capable of trimming this configuration well beyond the stall angle of attack. Although the pitching moment coefficient is shown to have a stable trend with angle of attack, some earlier test configurations did not. It was found that body-wing interference flow could cause neutral stability at certain combinations of angle of attack and elevon deflection.

Lateral-directional stability characteristics could be expected to be erratic above the stall angle of attack. The best to be hoped for is an aerodynamically stable configuration at angles up to the maximum lift angle. The delta-wing with its feature of maintaining attached flow along its central region assists greatly in attaining this goal.

A typical linearized directional stability characteristic is shown in Figure 6. The qualitative shape of this curve cannot be compared directly with data more commonly known to the reader, since this is a body axes presentation rather than the better-known stability axes. Even considering this, positive stability exists up to the maximum lift angle. The stability becomes increasingly negative at higher angles of attack. However, it must be remembered that the dynamic pressure is also decreasing and the resulting absolute moment is small. (The effects of this will be discussed later in the section on control.)

Lateral stability of a delta-wing configuration is not a significant problem in transition but may be quite severe in conventional flight, as seen in Figure 7. The combination of swept leading edge, a high wing location, and a high vertical tail can produce undesirably high lateral stability at conventional flight angles of attack. High lateral stability in transition is useful in helping to maintain straight flight paths during the landing maneuver.

Dynamic simulations of vertical attitude VTOL airplanes made by Ryan actually utilized non-linear aerodynamic data wherever required. This data is available to nominal sideslip angles of ±15°. Dynamic derivatives were estimated for simulation purposes but generally are of secondary importance compared with the static derivatives in transition.

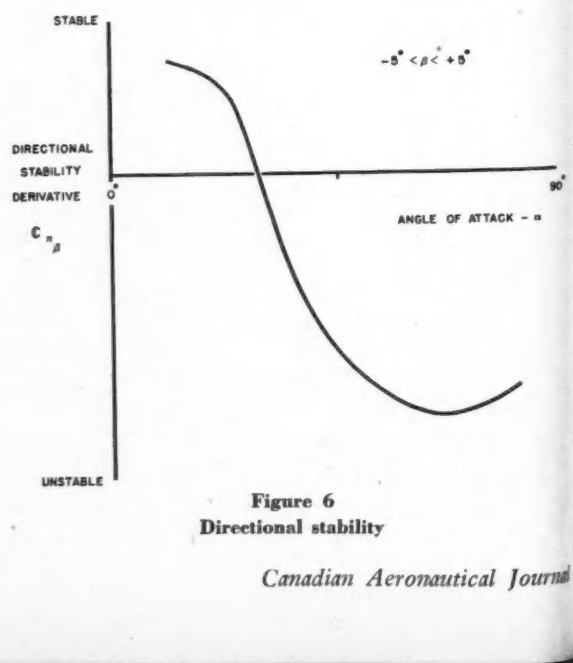

Figure 6
Directional stability

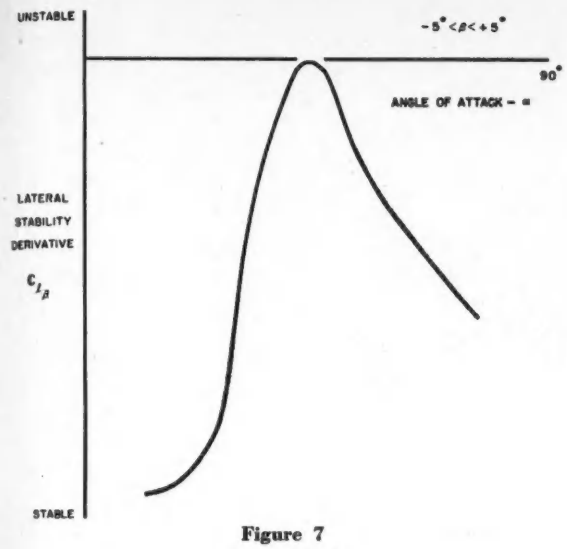

Figure 7
Lateral stability

CONTROL IN HOVERING

The following criteria are applicable to the design of reaction controls for hovering:

(1) dynamic balance,
(2) gyroscopic effects,
(3) gusts and steady state winds, and
(4) maneuverability.

A study of general hovering control requirements for VTOL aircraft was made in Reference 3 and compared with the helicopter flying qualities requirements of Reference 4. A correlation from this work is reproduced in part in Figure 8. Studies at Ryan have shown that high performance VTOL aircraft should have hovering control accelerations near those required of the helicopter. Although it may be possible to control a hovering vehicle with smaller accelerations than those required of the helicopter, the resulting maneuverability could be marginal.

Reference 4 also requires that a usable margin of control remains for acceleration even when the machine is hovering against 20 kt winds from the most critical azimuth angle. In the case of the delta-wing vertical attitude VTOL airplane, this critical angle occurs approximately 60° off the plane of symmetry, as shown in Figure 9. The roll power available from compressor bleed wing-tip roll ducts is adequate to maneuver the aircraft in hovering with the critical wind vector existing. It appears that the requirements for hovering this type of aircraft under adverse wind conditions can be met with little difficulty.

CONTROL IN TRANSITION

Criteria for establishing control requirements in transition are based on a combination of balancing and maneuvering the airplane. Balance is considered the problem of maintaining a given attitude within some acceptable tolerance limits. It is defined by the magnitude of the small motion perturbations from a mean attitude. The second criteria, that of maneuvering, makes the VTOL aircraft useful. It establishes the capability of the airplane to fly through transition, to be maneuvered into critical landing areas, and to operate under other than favourable wind conditions.

To the pilot, the controllability of the airplane in transition and hovering should follow the same instinctive modes common to conventional flight. This requires that the response of the airplane to a control input be in the same sense as in conventional flight with a similar response. Further, the cockpit control functions should be similar in the physical sense to the conventional controls. This is not necessarily a straight-forward problem of combining reaction jets with aerodynamic surfaces. However, experience gained from jet VTOL research has shown that the two control systems can be suitably mated.

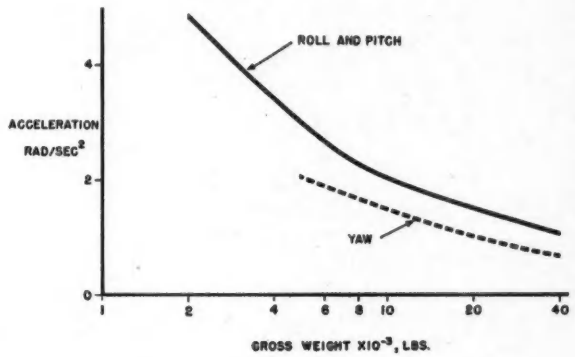

Figure 8
Helicopter control correlation

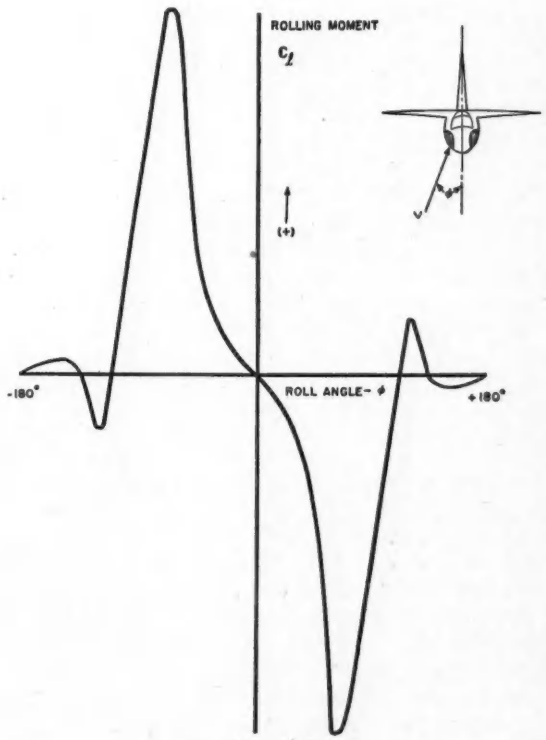

Figure 9
Rolling moment in hovering

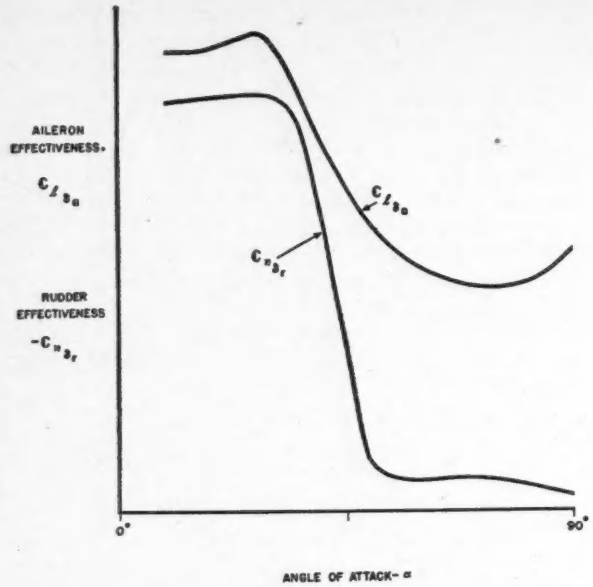

Figure 10
Aerodynamic control effectiveness

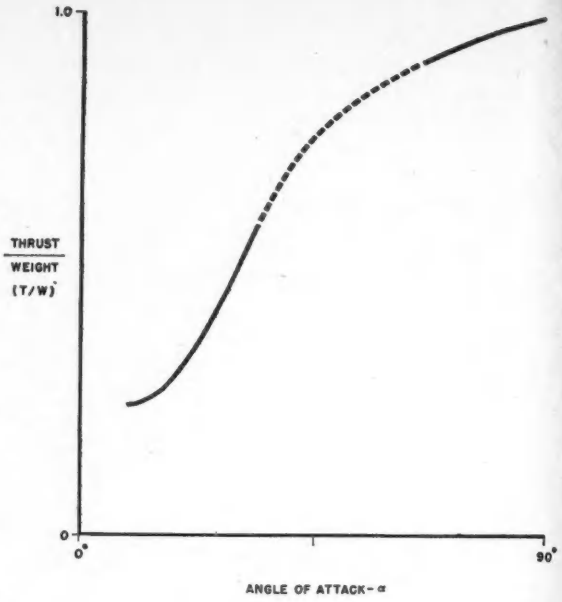

Figure 11
Thrust required in transition

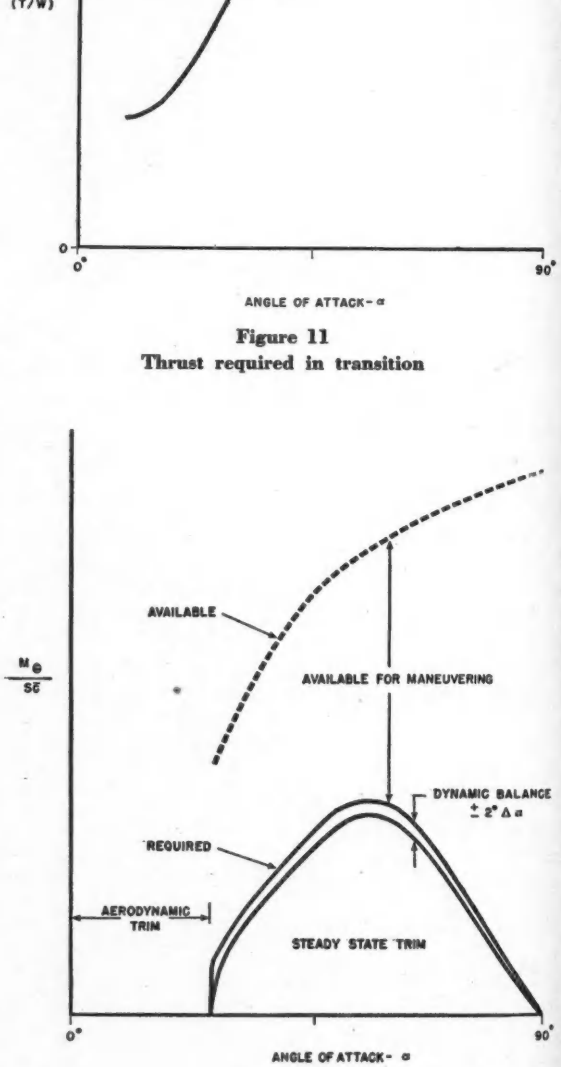

Figure 12
Longitudinal control in transition

All aerodynamic control surfaces on a delta-wing configuration are effective to relatively large angles of attack. As was shown in Figure 5, the elevator is capable of trimming the airplane well beyond the stall angle of attack and actually remains somewhat effective up through 90°. We have shown in Figure 10 control effectiveness curves for the aileron and the rudder. Adequate rudder effectiveness is available through 40° angle of attack while the ailerons remain effective up to 90°. As long as there is any airflow, the ailerons are fully capable of countering implied aerodynamic forces and moments. However, like all aerodynamic surfaces, they are ineffective in the pure hovering mode.

On a delta-wing design, control of the rudder can be by conventional direct linkage because of favorable hinge moment characteristics. However, in the case of the elevons, hinge moment reversal occurs under certain combinations of angle of attack and deflection, thereby requiring an irreversible actuating system.

In order to provide control at the lower air speeds, a system of jet reaction controls must be used. Pitch and yaw control can be obtained by deflection of the main propulsion jet. Roll control can be obtained from roll ducts located at the wing-tips and using compressor bleed air. Swirling the main jet stream may appear attractive for roll control but it has considerable design complexity and significantly reduces lifting thrust available.

In using the primary thrust for control purposes, the matter of transient thrust level becomes important. The thrust required to supplement aerodynamic lift varies with angle of attack approximately as shown in Figure 11. The relatively steep gradient in the vicinity of the maximum lift angle is characteristic of operating on the back side of the power required curve. This very feature provides an inherent thrust level adequate for the jet reaction controls which are required to supplement the aerodynamic controls in the transition region. In hovering and at low translational speeds, nearly full thrust is available for control purposes.

A complete picture of the control required in transition can be obtained by studying the criteria for control about each of the three axes. We have shown in Figure 12 the control requirements in the longitudinal mode. We are assuming that the aircraft is trimmed

aerodynamically before entry into transition and that all additional control to lower speeds is obtained through the jet reaction system alone. Three elements of the longitudinal control criteria are shown here: steady state trim, dynamic balance for a $\pm 2° \Delta\alpha$, and the excess control available for maneuvering. It is apparent that the requirements for dynamic balance are almost negligible and that the steady state trim does not strain the available pitch control. Therefore, a margin of control is available for changing attitude.

A similar breakdown of the directional control criteria is shown in Figure 13. We assume here that the airplane holds a fixed heading during the maneuver, thereby not requiring significant directional control. Dynamic balance is shown based on transient sideslip angles of $\pm 2°$. Separate control is allowed for compensation of the gyroscopic precessional moments of the engine rotor. After subtracting the dynamic balance and gyro compensation control requirements from the maximum available, there is still considerable control left for maneuvering if desired.

The only roll control criteria for transition is that associated with dynamic balance resulting from the sideslip angles previously discussed which leaves a large margin of roll power available for maneuvering as shown by Figure 14.

Contrary to some popular opinion, the vertical attitude VTOL aircraft is not an aerodynamic terror in transition. Although contemporary aircraft cannot safely fly the post-stall region, the vertical attitude VTOL has been designed to do it with safety. The previous stability and control data, although brief, show some of the reasons why this can be done and represent an advance in aerodynamic knowledge which cannot be overlooked.

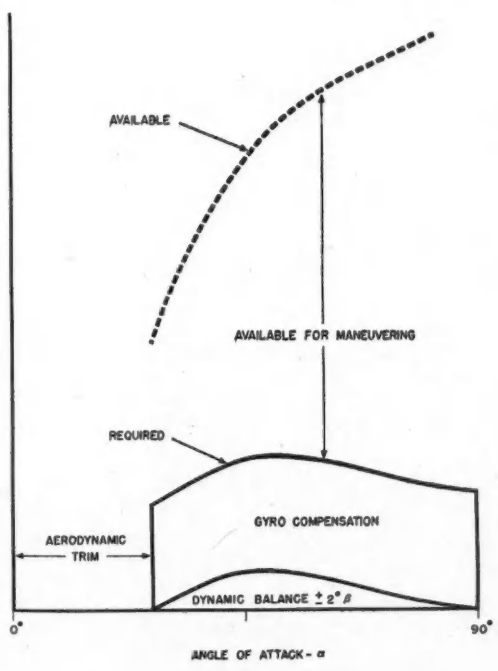

Figure 13
Directional control in transition

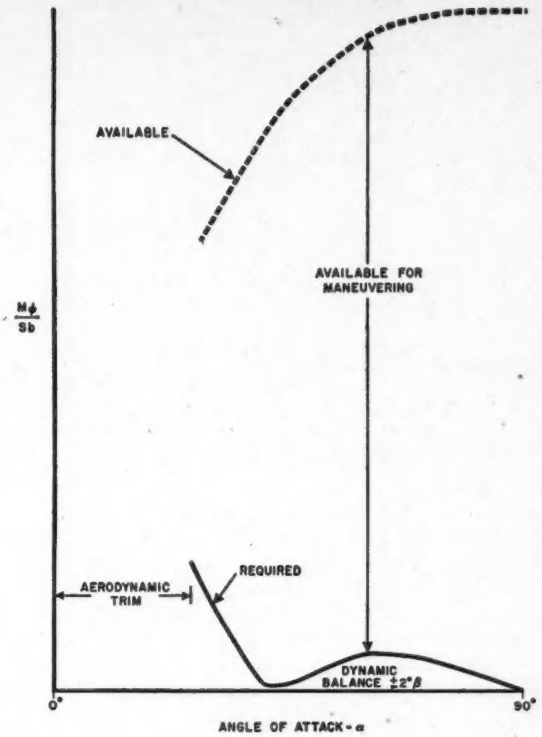

Figure 14
Lateral control in transition

ENGINE/AIRFRAME COUPLING

One of the lesser problems in past aircraft design which must be given consideration in VTOL airplane design is the effect of engine angular momentum on the coupling of the airframe pitch and yaw motions. Although this characteristic was known as far back as World War I[a], its importance had gradually disappeared with time, and only recently did it become of significance again.

During a study of methods of frequency response testing of turbojet aircraft[5], it was noted that there was significant coupling of the pitch and yaw motions of the F-86D aircraft. Although this was apparent in the flight test data, no significant observations of it have been made by pilots of this much-used aircraft.

It had always been assumed that engine gyroscopic moments might be significant in the transition and hovering modes. However, it was not until complete six degrees of freedom analog simulations were made of the conventional flight regime that this coupling effect was noted for conventional flight. Simulator tests of tricycle landing gear take-offs and conventional flight showed that a lightweight, high-thrust research aircraft would be affected by the engine gyroscopic characteristics. A typical example of this effect is the cross-coupling found in studies of a rapid tricycle landing gear take-off

[a]Page, V. W., Aviation Engines, Norman W. Henley Publishing Co., 1917, p 108: ". . . the gyroscopic effect of the rotary motor is detrimental to the best working of the aeroplane . . ."

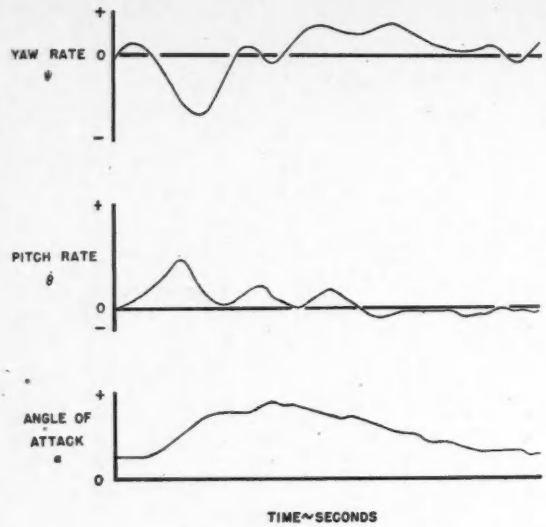

Figure 15
Horizontal take-off with high pitch rate

shown in Figure 15. The aircraft at high engine rpm was lifted rapidly from the runway to induce a maximum lift, minimum speed take-off. A high pitch rate was realized, which resulted in a high yaw rate. Both the pitch and yaw oscillations damped out quickly while the angle of attack continued well beyond maximum lift which would be dangerous with any other aircraft design.

This strong engine gyro coupling is not indicative of what would be found in an operational aircraft with VTOL capabilities. A comparison is made in Table 1 of airframe/engine gyroscopic data for several pertinent aircraft designs. The hypothetical VTOL aircraft is seen to have engine coupling parameters somewhat larger than current high performance aircraft.

Paiewonsky[6] has shown that negligible coupling exists when this parameter is below approximately 1.0. Some simulator studies of a hypothetical VTOL aircraft have been made at Ryan which confirm this result. Sample time histories from these studies are shown in Figure 16 for a particular configuration. Rudder kicks yielding sideslip angles of 2° were used as the forcing function. It is seen that the coupling of yaw rate into pitch rate differs little from the case of no engine coupling. The expected dissymmetry of coupling, due to direction of engine rotation, is also apparent. These results are typical of what was found throughout the simulator studies for both longitudinal and directional forcing functions over a wide range of Mach numbers.

Referring again to the data of Table 1, it could be expected that the engine coupling effects of an operational VTOL airplane would produce airplane handling qualities only slightly different from those of the P-51 airplane. However, the airframe coupling would be considerably different, generally following that of current high-performance aircraft incorporating the long, lean look.

THRUST CONTROL

Of particular importance in any VTOL flight is the matter of thrust control. There are two phases to this

TABLE 1
AIRFRAME/ENGINE GYROSCOPIC COMPARISON

Aircraft		F4B-2	P-51	F-86	Interceptor	Hypothetical
Year		1928	1943	1950	Current	Future
(T/W) S.L. Static		.28	.37	.42	.58	>1.0
Airframe Coupling	$\frac{I_z - I_x}{I_y}$	.73	.45	.92	.96	~.95
	$\frac{I_y - I_x}{I_z}$	.31	.28	.43	.64	~.7
Engine Coupling	$\frac{I_{xe}\omega_e}{I_y}$	.61	.71	.56	.14	~.7
	$\frac{I_{xe}\omega_e}{I_z}$	.46	.63	.43	.13	~.6

I_x, I_y, I_z: Moments of inertia about X, Y, Z body axes.
I_{xe} : Polar moment of inertia of rotating engine parts.
ω_e : Angular speed of rotating engine parts.

problem concerned with (1) maneuvering and (2) balancing the aircraft.

Maneuvering control is the conventional control used for establishing a desired nominal thrust level, which in VTOL operation governs lift as well as speed. The engine for VTOL propulsion, although selected primarily on the basis of thrust-to-weight ratio, must also have a favorable thrust response characteristic, as will be shown later. The basic thrust characteristic curve of a turbojet engine is shown in Figure 17. This is a common static thrust curve and shows the usual characteristic of a high thrust gradient at the higher rotational speeds. A vernier type power lever is adequate to provide the pilot with a means for precise control in this high thrust gradient region.

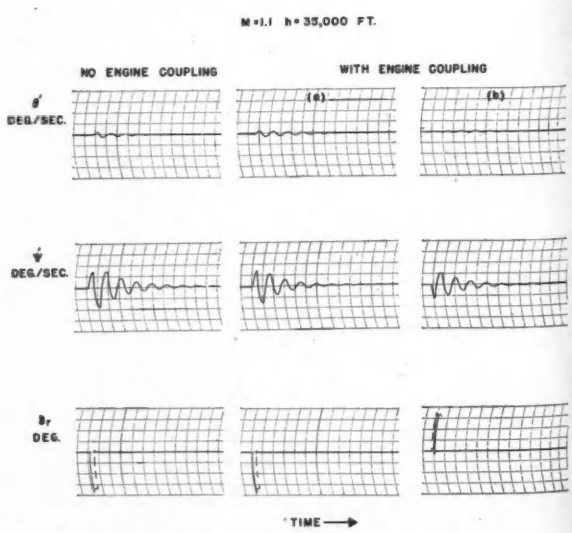

Figure 16
Gyroscopic coupling of hypothetical high performance aircraft

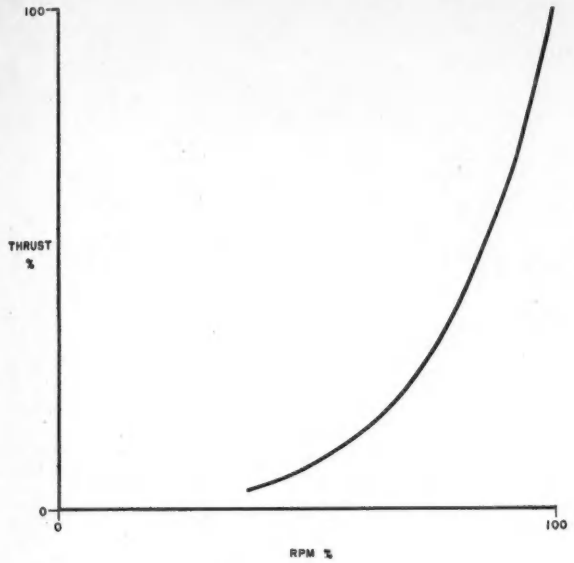

Figure 17
Thrust characteristics of turbojet engine

In flying through the landing transition, the pilot must coordinate pitch rate with throttle movement in order to assure a smooth transfer of the lifting force from aerodynamic to jet means. The drop-off in available lift with angle of attack, which was shown in Figure 4, is countered by an increase in engine thrust in the manner shown in Figure 18. The initial decrease in thrust occurs during the period of maximum lift and helps minimize any tendency to climb. The rapid thrust increase following this, while primarily picking up the lift, is also increasing the thrust available for jet reaction control. The thrust must finally be stabilized at a level equal to the airplane weight for the hovering mode. This thrust coordination has proved to be relatively easy to learn, and simulator "pilots" have mastered the technique with relative ease.

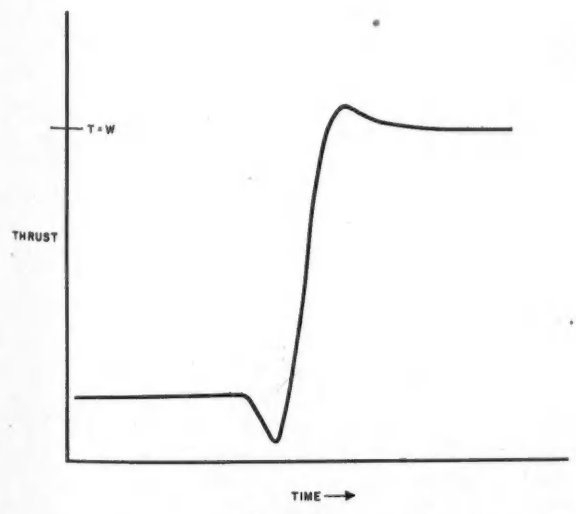

Figure 18
Pilot's thrust command during landing transition

The question arises as to the capability of an engine to accelerate (in thrust) as rapidly as shown in Figure 18. A typical engine acceleration curve is that shown in Figure 19. The time scale is equivalent to that of the previous figure. Changing the thrust level of this engine requires a change in rpm and tests have shown the engine is fully capable of giving the necessary thrust response.

The second phase of thrust control — that of balancing — is peculiar to the hovering mode. During landing and hovering operations, the pilot establishes a desired rate of descent through use of the vernier throttle. To supplement his efforts a simple thrust stabilization system might be used which can improve the smoothness of operation, and help prevent "bouncing" of the aircraft during thrust command changes. This system would operate through a limited thrust range of approximately ±5% at the hovering thrust level. A block diagram of such a thrust control system is shown in Figure 20. The controlling function is the longitudinal (vertical) velocity which is summed with the pilot's command to produce the desired instantaneous thrust level. This simple system could be used in controlling descent velocity and holding altitude in

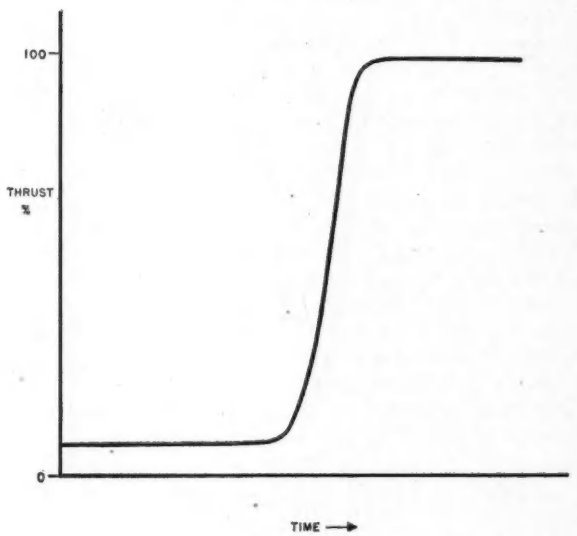

Figure 19
Thrust response of turbojet engine

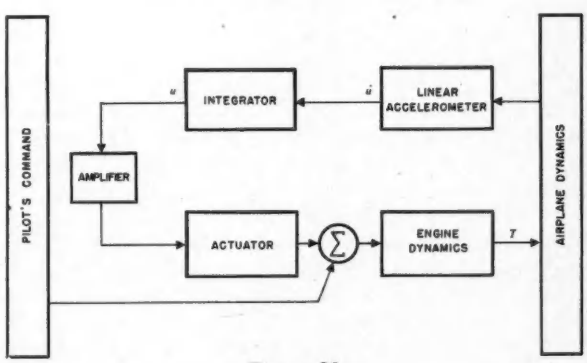

Figure 20
Thrust control system

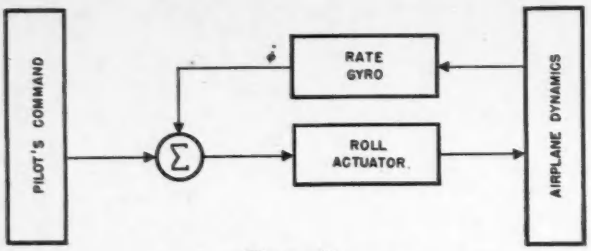

Figure 21
VTOL stability augmentation system

hovering. A description of a thrust-velocity control system is contained in Reference 7.

Many simulator tests and hovering flights have shown that a pilot can satisfactorily control the hovering aircraft safely without the use of thrust stabilization. However, the task of altitude control under varying ambient atmospheric conditions could be greatly simplified through the use of it.

STABILITY AUGMENTATION

We have previously discussed the aerodynamics of a delta-wing airplane as they exist through 90° angle of attack. Certain aspects of these aerodynamics might require some artificial improvement to assure suitable flying qualities. Zimmerman[8] has reviewed the stability requirements for vertically rising aircraft and has shown that damping may be a primary requirement in hovering. Scale model flight tests of a jet-powered delta-wing vertically rising airplane were made by the NACA Langley Field Laboratory to study the problems of transition flight. They were reported in Reference 9. After numerous transition flights in the Langley Full Scale Wind Tunnel, they concluded that the addition of roll damping augmentation made the model satisfactorily flyable throughout the entire speed range. As was mentioned previously, lateral stability is an important parameter and actually might overshadow the effects of negative directional stability. A simple stability augmentation system based on NACA studies might consist only of roll rate damping. A block diagram of such a system is shown in Figure 21.

A more complete stability augmentation system, which provides rate damping in transition and cross-coupling control and damping in hovering, is shown in Figure 22. This is a block diagram of a typical three-axes stability augmentation system. The French establishment of SNECMA has studied such a system for their "Flying Atar". Reference 10 gives a brief description of their system which utilizes the basic elements of rate damping and attitude reference. Cross coupling of signals between the pitch and yaw axes is provided to give gyroscopic precessional moment compensation in the hovering mode. The system shown in Figure 22 represents an attitude stabilization system which maintains the aircraft attitude in space as commanded by the pilot. Deviations from the pilot-selected attitude are countered by automatic error plus rate damping commands. Removal of the attitude sensors results in a rate stabilized system wherein the pilot commands angular rates and the augmentation system then holds those rates. Such a system may be entirely adequate for some applications. In addition to attitude stabilization about

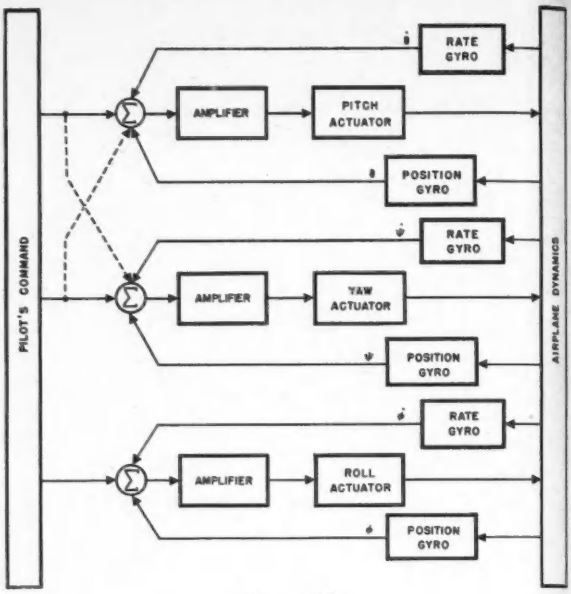

Figure 22
Complete stability augmentation system

the three axes, a thrust control system may be used for hovering. The thrust control system of Figure 20 and the more elaborate SNECMA system both stabilize the hovering aircraft along its vertical axes in hovering.

There have been many comments, pro and con, on the use of stability augmentation for transition and hovering purposes in VTOL aircraft. Some French, British and American airplanes and test vehicles using stability augmentation have demonstrated controllability and stability superior to that possible when using only the human servomechanism. The arguments against the use of stability augmentation assume that it requires considerable complexity of the aircraft which incorporates it. This assumption is completely unfounded when we compare it with even our current high performance aircraft which will be contemporary with tactical VTOL aircraft.

For a comparison, let us take a look at the automatic flight control system presently used in the current F-100D aircraft. A simplified block diagram of this system from Reference 11 is shown in Figure 23. Comparing the elements of the stability augmentation system, Figure 22, with those of the automatic flight control system of the F-100D in Figure 23, we immediately see that a number of components are common to both systems. There are 3 rate gyros, 3 angular attitude references, 3 amplifiers, and 3 control actuators. With minimum alteration of this conventional automatic flight control system, a complete VTOL stability augmentation system could be provided. Primarily, this modification would mean the use of components with wider operating ranges and perhaps the addition of the hovering thrust control. Automatic gain changing is already present in the conventional system and could easily be adapted to include transition and hovering if necessary. The operational vertical attitude VTOL aircraft automatic control system would simply be an integration of the basic elements of Figures 22 and 23.

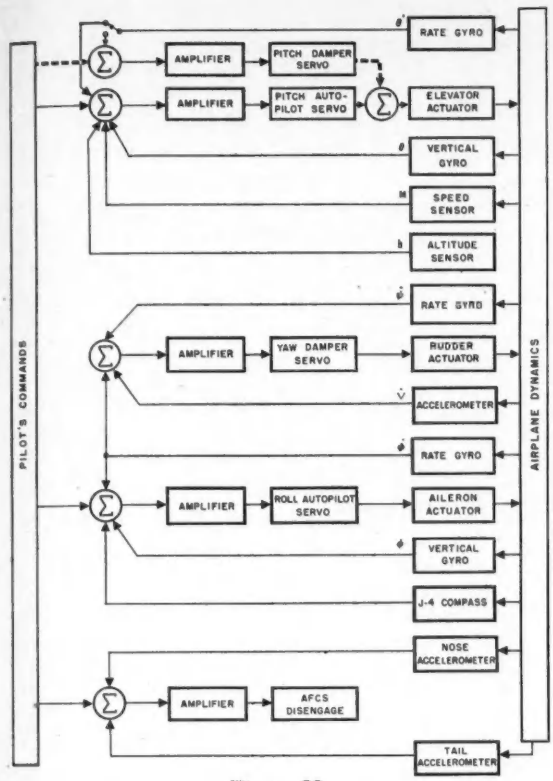

Figure 23
MB-3 automatic flight control system

Reliability of operation of an integrated system would be high since it is active during the entire flight, thereby eliminating failures which are due to re-energizing the system in flight. It also eliminates the need for checkout during flight when the pilot is occupied with navigating to his target. Reliability of this system (as with any system) is further assured through selection of components suited to the system and the use of latest concepts of redundant and parallel control loops. The overall state of the art of circuit design and component development is rapidly advancing because of the great dependence of missile success on their development. Continued improvements in reliability of automatic systems by the aeronautical and electronic industries will also help assure the reliability of the VTOL control system. From the basic philosophy of VTOL design, i.e. to eliminate all dead weight items from the airplane that are not required for airborne flight, we can formulate a second philosophy of making maximum use of all airborne systems to satisfactorily attain a high degree of weapon system usefulness.

FLIGHT CHARACTERISTICS

The ease with which a delta-wing vertical attitude VTOL airplane can accomplish transition flight is graphically illustrated in Figures 24 through 26. Figure 24 shows a time history of transition from hovering to conventional flight. The aircraft is initially translating forward in the hovering mode at a low air speed. The pitching moment from the positive pitch jet deflection causes the aircraft's nose to pitch forward, and the aircraft then tends to accelerate to conventional flying speed. The smoothness of aircraft control is indicated by the small deviations of the yaw and roll jet from their neutral positions. The total transition time from start of pushover until aerodynamic flying speed is reached would be approximately equal to a fast horizontal take-off.

A transition from conventional flight to hovering flight is shown in Figure 25. The nose of the airplane is raised by introducing negative pitch jet deflection. As the pitch attitude angle increases, the negative jet de-

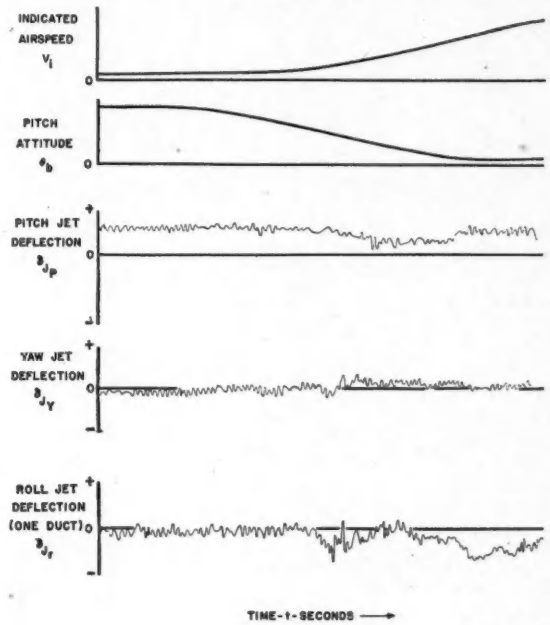

Figure 24
Take-off transition time history

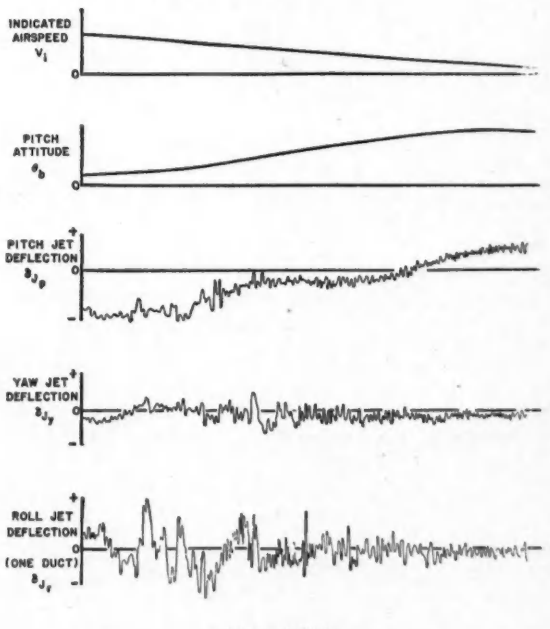

Figure 25
Landing transition time history

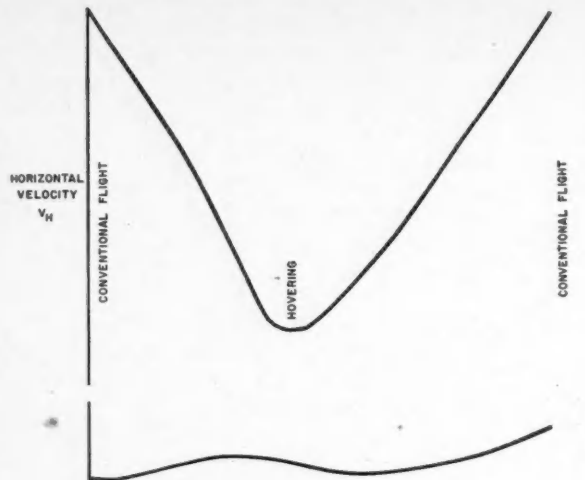

Figure 26
Altitude variation in transition

flection is reduced, and as the pitch attitude approaches 90°, positive values of jet deflection tend to reduce the pitching rate. The airplane finally comes to a hovering attitude with a low translational speed. The action of the stability augmentation system about the three axes appears to be somewhat greater than during a take-off transition. However, it is relatively small and represents only a small part of the control power available. The greater activity of the reaction jets in stabilizing the airplane is noted in the immediate post-stall region of the wing and decreases markedly as the aircraft approaches the hovering attitude. (The pitch attitude curves are approximately equal to the angle of attack since only small altitude changes are involved during the transition maneuver.)

A plot of the altitude variation during a transition is shown in Figure 26. This transition at a low altitude shows a minor altitude change for a simulated landing and a corresponding simulated take-off transition altitude change only slightly greater.

The successful flights of the X-13 airplane have amply demonstrated that transitions are easily made near the ground over varying types of terrain. The maneuverability feature is an important aspect of operational VTOL airplanes and would be facilitated by a stability augmentation system.

CONCLUSIONS

As a result of four years of development effort on vertical attitude jet VTOL airplanes and many other associated studies, it can be concluded that:

(1) Flight beyond the angle of wing stall is neither impossible nor impracticable and can be safely accomplished based on today's knowledge.

(2) Aerodynamic and jet reaction controls can be designed to work together in complete harmony and provide satisfactory control in hovering, transition and conventional flight.

(3) Reaction controls must be designed to yield sufficient control power to simultaneously accomplish the functions of
  (a) balancing the vehicle,
  (b) compensating for gusts and steady winds, and
  (c) providing a useful degree of maneuverability.

(4) Engine gyroscopic effects can be adequately compensated in hovering through a simple cross-coupling of pitch and yaw commands and will not be significant in conventional flight for operational VTOL aircraft.

(5) Conventional turbojet engines have adequate dynamic response for hovering which can be further supplemented by a simple longitudinal stabilizing system.

(6) Stability augmentation for transition and hovering can be provided by a system which is also used for the conventional flight regime and fire control purposes.

(7) Transition and hovering flight is readily accomplished with the proper integration of pilot, airframe and control system.

(8) The vertical attitude VTOL airplane is a proven feasible design whose concept can be incorporated into high performance aircraft.

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RECENT ADVANCES IN THE AERODYNAMIC DESIGN OF AXIAL TURBOMACHINERY†

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SUMMARY

The requirements for compressors and turbines for turbojet engines are that they be light, reliable and efficient. If reliability is a function of the stress level imposed on the component, the stress level may be set by considerations of materials, operating temperature and required life of the component. For a given stress level then, the aerodynamic loading of the compressor or turbine, in terms of relative Mach number, axial Mach number, blade loading and so forth, should be maintained at a high level to minimize the size and weight of the component. If these highly loaded blades can be designed with loss coefficients compatible with the higher work levels, highly loaded compressors and turbines of reasonable efficiency should result.

By utilization of transonic rotor tip relative Mach numbers and low levels of inlet whirl in conjunction with proper loading limits, high-flow high-pressure-ratio compressor stages are possible. The feasibility of such designs was demonstrated by a single stage which had a relative Mach number at the tip of 1.1. A stage pressure ratio of 1.48 was obtained with a maximum airflow per unit frontal area of 30 pounds per second per square foot and a peak efficiency of approximately 0.9.

In order to design similar stages to meet various design requirements of flow and pressure ratio, a correlation of losses with blade loading is necessary. A diffusion parameter with which losses have been correlated has been conceived and successfully applied. The diffusion parameter can be computed solely from the vector diagrams and is, therefore, easily applicable to current design procedures. The correlation indicates values of the diffusion parameter above which the losses significantly increase, and thus limits for a particular design can be easily determined. By proper application of transonic design and the loss-diffusion correlation, efficient stages can be designed for high flow rates and high pressure ratios.

High-mass-flow high-work turbines can be designed to operate satisfactorily over a range of work levels if the design point is chosen in proper relation to limiting blade loading for the last turbine rotor. Satisfactory turbine efficiency can be obtained if the blade loading and blade-surface diffusion are taken into consideration in the design of the blade profile. The velocity distribution of the blade profile can be analytically determined through the use of a blade-to-blade stream-filament solution in conjunction with an axisymmetric solution in the radial-axial plane.

INTRODUCTION

THE requirements for compressors and turbines for turbojet engines are that they be light, reliable and efficient. These requirements have dictated the need for high-capacity continuous-flow compressors and turbines with high mass flow per unit frontal area, high pressure ratio and high work output per stage, high efficiency and broad operating range.

High values of airflow capacity and work per stage are associated with high flow Mach numbers and high values of blade loading (blade lift). Without a proper blade loading criterion, the losses associated with high Mach number blading can be excessive. The requirement of good component efficiency, therefore, led to compressors and turbines that were designed for low relative Mach numbers and associated low values of airflow, pressure ratio and work output. The turbojet engines incorporating component designs of this type were identified with high values of specific weight.

Considerable effort has been directed toward component design of systems that utilize high relative Mach numbers and high blade loadings effectively. This paper discusses such a system. First, the general component-design considerations are discussed. Second, theoretical component-design parameters and techniques that lead to high flow and work levels with good efficiencies are presented.

PRELIMINARY DESIGN CONSIDERATIONS

The compressor and turbine designer is faced with satisfying several engine cycle requirements. The component-design-point conditions (pressure ratio, work output, airflow) must be obtained with good efficiency and the off-design performance of the components must be satisfactory. The final component design results from many compromises that must be made in each step of the design procedure. This section discusses the preliminary design requirements, considerations and compromises.

Reliability

Since turbojet engines must be structurally sound, the aerodynamic and mechanical design of the components must be consistent with the allowable material stresses. If the reliability of the components is a function of the operating stress level, the material stresses should, from a mechanical design standpoint, be kept low. In contrast, from an aerodynamic point of view with the trend toward higher mass flows, high turbine-inlet temperatures and higher tip speeds, it is desirable to operate close to the material stress limits in order to achieve minimum weight. Many variables covering a wide scope must be considered in a detailed component stress analysis. As a simplified approach, however, if only centrifugal stresses (S_c) are considered, the compressor or turbine stress level is mainly a function of

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rotative speed and annulus area

$$S_c \sim \omega^2 A$$

or

$$S_c \sim U_t^2 \left[1 - \left(\frac{D_h}{D_t}\right)^2\right]$$

(Reference 1) where ω is rotation speed, A is annulus area, U_t is compressor tip speed, and D_h/D_t is the hub-to-tip diameter ratio. Therefore, in the selection of rotative speed and annulus area, the aerodynamicist must be continually guided by the mechanical aspects of the problem. Component stress analysis is not considered in detail in this paper. However, stress considerations underlie the succeeding discussion even though they are not specifically presented.

Component weight

One of the primary considerations in component design for aircraft powerplants is weight. It is of course desirable to minimize the component weight in order to make the powerplant and flight vehicle lighter and more compact. If the engine weight is reflected in the diameter and number of stages of the compressor and turbine, high airflow per unit frontal area, high compressor pressure ratio per stage and high turbine work per stage are goals to be achieved to yield minimum component diameter and length.

Unfortunately, these goals are not always compatible. In the compressor, for example, high airflow requires low hub-tip diameter ratios. For a given stress level, a reduction in diameter ratio means a reduction in blade tip speed. Within specified aerodynamic limits, this reduction in tip speed results in a reduction in pressure ratio per stage and, consequently, in an increase in the number of stages. Thus, the designer is faced with the choice of a large-diameter short compressor or a small-diameter longer machine. The question of which unit affords the best compromise in the final aircraft installation can only be answered by a detailed mechanical analysis of each problem, which would be beyond the scope of this paper. In either case, however, high compressor pressure ratio per stage and high turbine work per stage provide a reduction in component weight.

Efficiency

Inasmuch as the specific fuel consumption is directly affected by component efficiency, the designer is constantly striving for a high ratio of work output to losses. Therefore, the designer must select both design velocity diagrams and compressor blading carefully to avoid blade stall and blade-row choking, since these conditions are major sources of loss. This choice must be guided by intelligent use of blade loading parameters and Mach number working levels. In addition, the compressor and turbine should be matched so that the components operate at peak efficiency simultaneously at the flight condition at which these requirements are considered most critical.

Off-design performance

Turbojet engines operate over a wide range of flight conditions. Therefore, the compressor and turbine are required to have satisfactory performance not only at the design point but also over a fairly wide range of speeds and airflows. For example, when the engine accelerates rapidly from idle speed to full power, compressor surge, combustor flameout and turbine overtemperature must be avoided. The components must have a reasonable overspeed margin (for operation with maximum thrust at subsonic flight Mach numbers) and must operate satisfactorily at low equivalent speeds (corresponding to supersonic flight Mach numbers). These operating conditions, as well as the design-point condition, must be considered.

COMPONENT DESIGN

As stated previously, there are three primary component-design objectives: (1) high airflow per unit frontal area, (2) high pressure ratio per compressor stage and high work output per turbine stage, and (3) high efficiency. The philosophy that is utilized in achieving these goals in conjunction with the variables involved is indicated in this section of the paper. It becomes obvious that the primary responsibility of the component designer lies in relating these objectives to design vector diagrams and in choosing blading to establish the velocity diagrams. Although the variables are discussed separately, they are interrelated, and optimizing one quantity usually results in a sacrifice to another.

Flow per unit frontal area

The mass flow per unit frontal area of a compressor is primarily a function of three variables: (1) hub-to-tip diameter ratio, (2) inlet axial Mach number, and (3) inlet flow angle.

A plot of equivalent weight flow per unit frontal area against inlet axial Mach number for various values of hub-to-tip diameter ratio is presented in Figure 1 for zero inlet flow angle. It is apparent from this figure that in order to achieve high airflow per unit frontal area both the annulus area and the axial Mach number should be a maximum.

Compressor designs which do not incorporate prerotation of the inlet flow have certain advantages in that a greater potential for high mass flow exists. This is illustrated in Figure 2, where airflow per square foot of annulus area is plotted against inlet flow angle for various axial Mach numbers. This plot indicates that at any axial Mach number the flow rate decreases as the inlet flow angle increases.

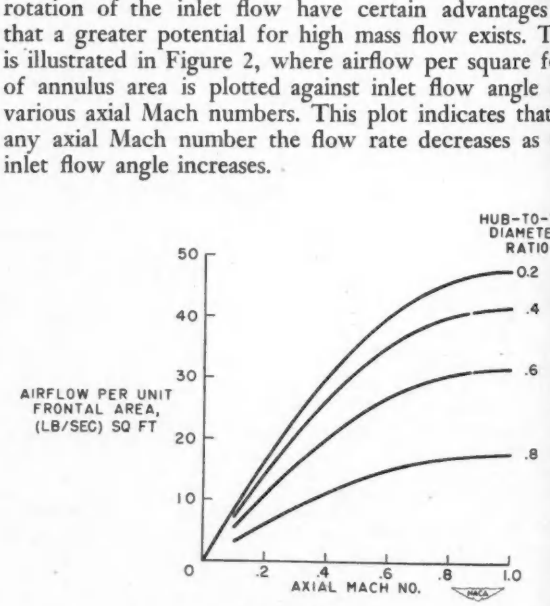

Figure 1
Airflow characteristics of turbomachinery

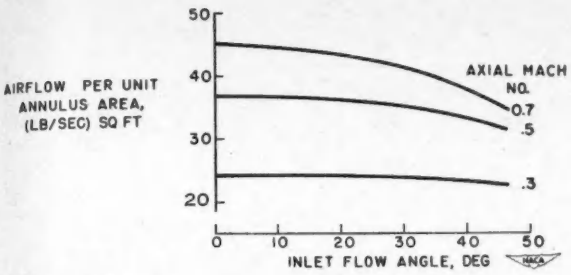

Figure 2
Variation of airflow rate with inlet flow angle

The hub-to-tip diameter ratio is limited primarily by mechanical considerations to a value of approximately 0.35. With hub-tip ratios below this value, blade mounting problems are difficult because the hub diameter is relatively small, and blade bending and vibratory stress often become severe because of the increased blade length. The maximum mass flow is obtained when the axial Mach number in the annulus is 1. However, the Mach number is limited primarily by choking considerations, not in the annulus but in the blade row itself. Blade-row choking occurs at an annulus Mach number less than 1. In order to avoid high losses usually associated with choking in the blade row, values of annulus axial Mach number are usually restricted to 0.7 or less.

Pressure ratio per stage

The energy input to a compressor stage is commonly expressed in equation form as

$$\Delta H = U\Delta V_\theta$$

where ΔH is the energy addition, U is the blade speed and ΔV_θ is the change in tangential velocity across the blade row. The pressure ratio is related to the energy input by the stage efficiency, and the change in tangential velocity is a function of the inlet Mach number and blade loading. Therefore, the primary variables which affect stage pressure ratio are: (1) blade speed, (2) inlet Mach number, (3) blade loading, and (4) efficiency.

The designer is limited in the choice of rotational velocity by component stress considerations. The normal working range of compressor tip speed is 1,000 to 1,100 fps. With this limitation on wheel speed, high pressure ratio must be achieved by utilizing high inlet Mach number and high blade loading. The effect of Mach number and blade loading on pressure ratio is indicated in Figure 3, where pressure ratio is plotted against relative Mach number for various values of blade loading. These curves were calculated for a compressor blade speed of 1,100 fps and an efficiency of 100%. It can be noted that pressure ratio increases with both inlet Mach number and blade loading. As stated previously, optimizing one compressor design parameter usually results in some sacrifice in another parameter. Therefore, these parameters must be examined in terms of their effect on efficiency.

The influence of Mach number on efficiency is illustrated in Figure 4. This curve was obtained from a fixed-geometry subsonic compressor operating over a range of speeds[2]. For conventional subsonic airfoil shapes, efficiency remains at a high level until a relative Mach number of about 0.7 or 0.8 is reached and then decreases

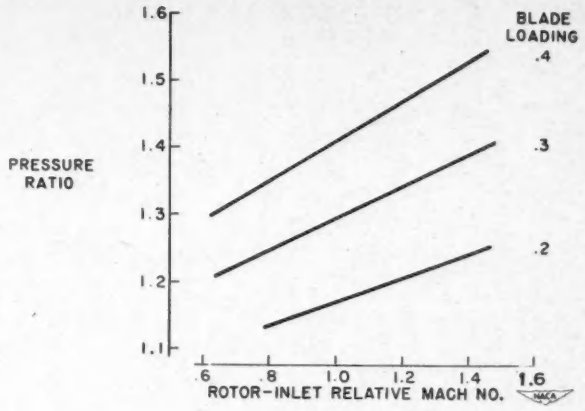

Figure 3
Effect of Mach number and blade loading on pressure ratio

rapidly. As a result, the compressor designer has usually sacrificed pressure ratio in order to stay within reasonable efficiency levels. Obviously, then, the problem confronting the aerodynamicist in his attempt to optimize a compressor is that of extending the Mach number range over which the compressor can operate efficiently.

The reduction in efficiency has been attributed to shock losses associated with supersonic velocities on the blade. However, these losses are not necessarily the sole cause, since the blade loading becomes greater, and possibly excessive, as the Mach number is increased. Therefore, it was suspected that the failure of high Mach number compressor designs was due to a combination of high Mach number and high blade loading. Considerable effort was devoted to finding a satisfactory blade loading parameter. The results of this work are published in Reference 3, where a blade loading parameter called the "diffusion factor" is presented. The basis of the development of the diffusion factor is shown in conjunction with a typical compressor velocity diagram in Figure 5. In general, at the design incidence angle, the velocity along the suction surface of a compressor blade attains a maximum value near the leading edge and then decreases to the level of the outlet velocity at the trailing edge. For high values of pressure rise, this velocity difference may become large and may result in separation of the boundary layer, a condition referred to as blade stall. This phenomenon is associated with large viscous flow losses. Since it is impracticable to compute

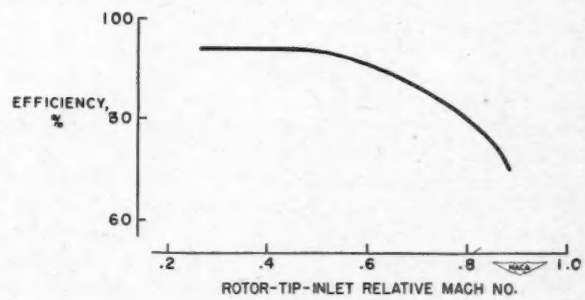

Figure 4
Variation of compressor efficiency with Mach number for conventional design

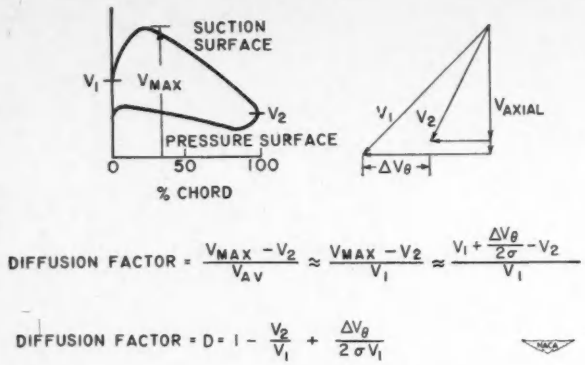

Figure 5

Development of diffusion factor *D*

suction-surface velocities for various blade shapes over wide ranges of design conditions, in the interest of simplicity it was desirable to obtain an approximate blade loading parameter. This is feasible because compressor blades are generally selected from families of airfoils, such as the 65 series or circular arc blade sections. Therefore, the general shapes of the velocity distributions around the blades are similar for any one series of airfoils. Specifically, the diffusion factor, by means of several simplifying approximations and assumptions, relates the maximum suction-surface-velocity difference of a typical compressor-blade velocity distribution at the design incidence angle with the over-all velocity characteristics and the geometry of the blade element. The diffusion factor D can be computed from velocity diagrams by the following relation:

$$D = 1 - \frac{V_2}{V_1} + \frac{\Delta V_\theta}{2\sigma V_1}$$

where V_2/V_1 is the relative velocity ratio across the blade row, ΔV_θ is the change in tangential velocity across the blade row, and σ is blade solidity, the ratio of blade chord to blade spacing.

The effect of blade loading on efficiency (as described by the diffusion factor) is shown in Figure 6. A rotor tip section loading limit appears to exist (values of D of 0.3 to 0.4), above which sacrifices in efficiency must be accepted.

Transonic compressors

The development of a reliable blade loading design parameter (the diffusion factor) was a significant step toward achieving high-performance compressors. By application of this theory, high-flow high-pressure-ratio compressors were designed and built at the NACA Lewis laboratory. The compressors were designed to operate at transonic relative inlet Mach numbers of the order of 1.0 to 1.1. The blade loading levels were chosen with the diffusion factor as a guide to obtaining good efficiency. In general, values of rotor tip diffusion factor were in the range 0.3 to 0.4. The over-all performance of the first transonic compressor stage is presented in Figure 7. (Design and performance details are presented in Reference 4.) The rotor operated without inlet guide vanes and the performance characteristics were very

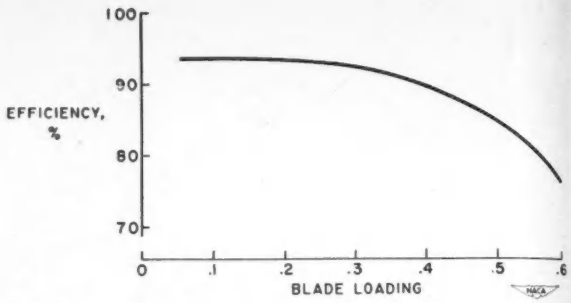

Figure 6

Effect of blade loading on efficiency of compressor rotor

good. A stage pressure ratio of 1.48 was obtained with a maximum airflow per unit frontal area of approximately 30 lb per second and a peak efficiency of about 0.9. This was an increase in airflow rate of about 20% over that for conventional subsonic compressors. In regard to pressure ratio, two subsonic stages would be required to produce the pressure ratio that this transonic unit produced in a single stage. In addition, the efficiency and range of efficient operation were comparable to those for subsonic compressors. The transonic compressor, therefore, represents a major advance in the compressor field in that it enables the designer to increase airflow capacity and stage pressure ratio simultaneously without sacrificing efficiency or range.

The performance data in Figure 8 show that the range of Mach number over which high efficiency can be maintained in transonic designs extends to values above a Mach number of 1. This results in substantial gains in weight flow per unit frontal area and in pressure ratio per stage over those of conventional subsonic compressors. These results were achieved by utilizing the proper combination of blade loading and Mach number in conjunction with the proper blading to establish the design velocity distributions.

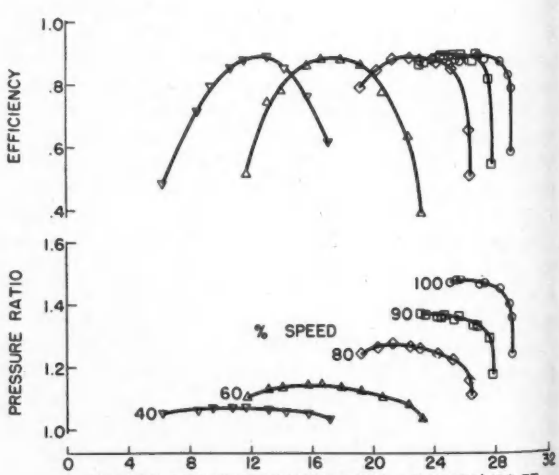

Figure 7

Performance of first transonic compressor

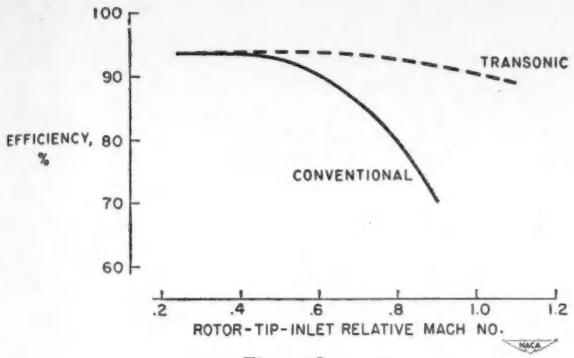

Figure 8

Variation of efficiency with Mach number for conventional and transonic compressors

The phase of the compressor design procedure that concerns the selection of blading is not discussed in detail in this paper but it has played an important role in the success of transonic compressors. The majority of transonic compressor blading was selected from a family of double-circular-arc airfoil sections. This airfoil shape is composed of circular-arc suction and pressure surfaces. High Mach number blading differs from conventional airfoils in that the leading edge of the blade is thin and the position of maximum thickness is shifted toward the trailing edge. Both modifications helped reduce the maximum surface velocity and minimized shock losses and boundary-layer separation. The design philosophy that was utilized in the choice of this blading is presented in detail in Reference 5. Experimental evidence of the successful application of double-circular-arc blading to compressor design is reported in Reference 6.

Turbine weight flow per unit frontal area

The fundamental criteria for proper aerodynamic design of compressors can be applied to turbines. The weight flow per unit frontal area is a function of a hub-tip diameter ratio and an axial Mach number (Figure 1), the exit diameter ratio and the exit axial Mach number.

Of course, the choice of a diameter ratio for turbomachinery involves a compromise between several desirable features, any one of which may be controlling for a given design. If high weight flow per unit frontal area is a prime requirement for the turbine, the minimum diameter ratio is usually set by mechanical considerations. Practical mechanical considerations of fixing closely spaced highly stressed blading to a disk usually limit the minimum diameter ratio to 0.5.

For any given diameter ratio, the airflow capacity of a turbine is determined by the axial Mach number at the exit of the turbine. This axial Mach number cannot exceed a Mach number corresponding to limiting blade loading for the last rotor. When the last stage of a turbine is operating at its limiting loading point, maximum turbine work is being produced for that blade speed. Any increase in turbine pressure ratio above that required to produce limiting loading in the last rotor does not result in any increase in turbine work. A simplified fundamental picture of the flow phenomena that accompany limiting blade loading[7] is presented in Figure 9. When the local axial Mach number within the last rotor blade row becomes sonic, any further decrease in pressure downstream of the rotor cannot be transmitted upstream to the rotor blades. Thus, an increase in turbine pressure ratio results in an increase in axial Mach number behind the turbine, without the increase in whirl velocity that must accompany greater work outputs. For typical blading, this flow condition corresponds to an axial Mach number in the annulus of about 0.7. Choice of a design exit axial Mach number near this limit would result in a turbine that had no range in work output above that required for design-point operation.

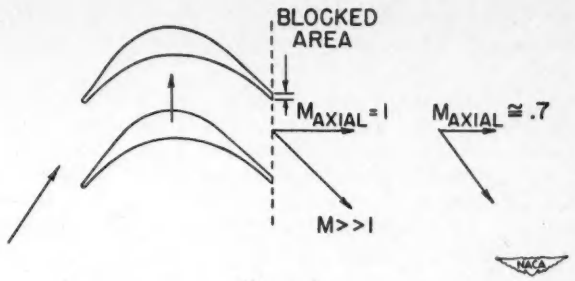

Figure 9

Limiting loading flow conditions in turbine rotor

Figure 10 presents a typical variation of turbine efficiency over a range of work output at constant blade speed. Turbine operation at pressure ratios above that required to produce limiting turbine work output is obviously a condition to be avoided. Turbine efficiency at limiting loading is typically lower than that at work outputs slightly less than the limiting value. The decrease in efficiency, as limiting work output is approached, is mainly due to the shock losses associated with the high rotor relative exit Mach numbers. For typical blading local surface Mach numbers of 1.5 can be encountered. Therefore, the attainment of airflow capacity through the use of high values of exit axial Mach number must be compromised for satisfactory operating efficiency as well as latitude in operating range.

Turbine work

With the rotational speed of the turbine set by compressor aerodynamics or component stress considerations, turbine work output per stage depends directly on flow velocity level. The higher losses that are associated with the higher velocities need not result in a corresponding decrease in efficiency, since the level of work output also rises. The decrease in efficiency that accompanies

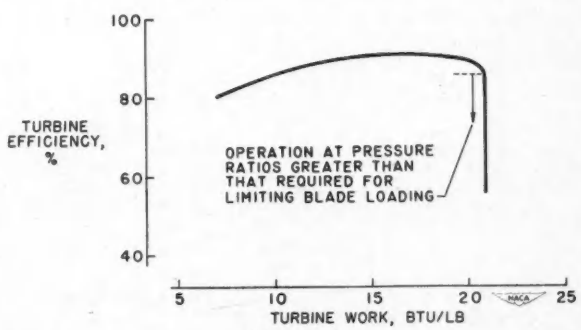

Figure 10

Typical variation of turbine efficiency with turbine work at constant blade speed

the higher work output can be minimized by careful consideration of the aerodynamic loading on the turbine blade.

The manner in which blade loading criteria can be applied to a turbine is fundamentally the same as the diffusion-factor approach used for compressors. The high blade solidities that are encountered in turbines do not preclude the use of the diffusion factor, but rather permit a more accurate evaluation of the blade-surface velocities and diffusion encountered in a particular design. A flow-channel type solution, such as the simplified stream-filament technique[8], has provided good agreement between predicted and observed surface velocity distributions on a turbine stator blade. Use of a method such as this in conjunction with an axisymmetric solution in the radial-axial plane yields the desired surface velocity distribution. The magnitudes of the surface velocities and the surface diffusion that result from such a solution are indicative of the suitability of that particular profile shape and blade solidity.

When conventional techniques are used in designing blades for high-work-output turbines, severe penalties are often paid in terms of efficiency. These design techniques often yield blading that will not produce design work. The fundamental reason is lack of recognition of proper loading requirements of the rotor blade row, as would be true in a corresponding case for the compressor. Figure 11 represents velocity distributions for conventionally designed turbine rotor blades. The high peak surface velocities and high surface diffusion of the high-work-output blading are indicative of a high-loss blade row. This poor flow condition can often cause additional losses in a turbine. In the simple case of a single-stage choked-rotor turbine, rotor losses not anticipated at the time of design will result in operation with lower-than-design stator-exit velocities. The accompanying reduction in work capability may be offset by an increase in the whirl energy at the exit of the rotor. Since high-work turbines operate near limiting work output, a further increase in loss is incurred at the exit of the blade row. If the turbine is designed to operate too near limiting work output, or the rotor losses are too high, or both, the turbine can reach limiting work output before design work is achieved.

Figure 12 illustrates the type of velocity distribution that can be obtained on turbine blading by careful design. While there is little or no improvement to be made on the blading of low-work-output turbines, a marked change in peak surface velocities, as well as diffusion, can be noted for the high-work-output blading. Consideration of the flow conditions in the channel formed

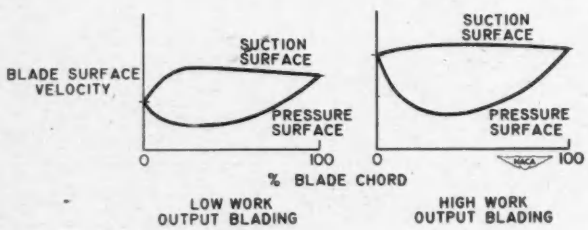

Figure 11
Typical rotor-blade-surface velocity distributions for conventional blading

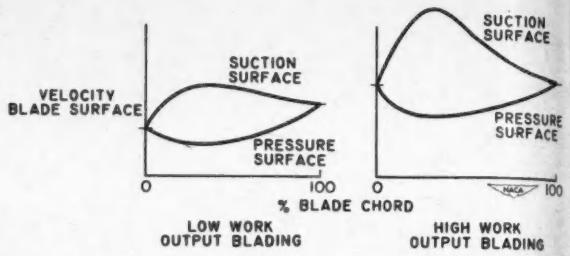

Figure 12
Typical rotor-blade-surface velocity distributions for blading designed by quasi-three-dimensional technique

by adjacent rotor blades resulted in selection of a blade solidity and profile that minimize the losses at high turbine work outputs.

CONCLUSION

There are three primary objectives of compressor or turbine design: (1) high airflow per unit frontal area, (2) high pressure ratio per compressor stage and high work output per turbine stage, and (3) high efficiency. These goals can be achieved by proper selection of Mach number and blade loading operating levels. For the compressor, for example, the range of rotor tip Mach number over which high efficiency can be maintained has been extended to values above 1. This resulted in substantial gains in weight flow per unit frontal area and in pressure ratio per stage over those of conventional subsonic compressors. The design of high-work-output high-mass-flow turbines that will operate at a reasonable level of efficiency requires proper selection of the turbine operating point in relation to limiting blade loading and consideration of the velocity and diffusion levels on the blade surfaces.

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(4) Lieblein, Seymour, Lewis, Jr., G. W., Sandercock, D. M. — *Experimental Investigation of an Axial-Flow Compressor Inlet Stage Operating at Transonic Relative Inlet Mach Numbers. I — Over-All Performance of Stage with Transonic Rotor and Subsonic Stators up to Rotor Relative Inlet Mach Number of 1.1*, NACA RM E52A24, 1952.

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(7) Hauser, C. H., and Plohr, H. W. — *Two-Dimensional Cascade Investigation of the Maximum Exit Tangential Velocity Component and Other Flow Conditions at the Exit of Several Turbine-Blade Designs at Supercritical Pressure Ratios*, NACA RM E51F12, 1951.

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TECHNICAL FORUM

Foreign Objects in Aircraft, Engines and Airborne Equipment

Figure 1

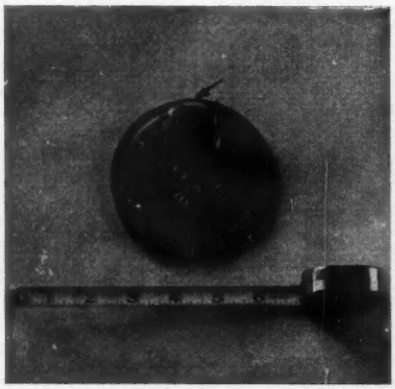


Figure 2

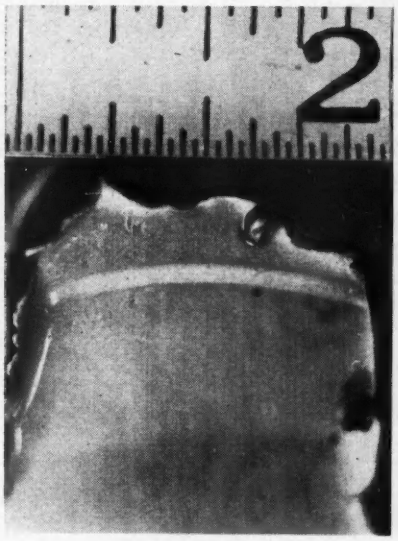


Figure 3

FURTHER to the discussions which took place during the "Foreign Objects" symposium, herewith several photographs which you might find of interest.

In the Engine Laboratory of the National Aeronautical Establishment, we have obtained a number of used lubricating oil cans from several service stations. These cans had been opened by the conventional method of impaling them on a combined funnel and opener.

Photographs were taken to show whether these cutters introduced undesirable particles into the oil.

Figure 1—shows the top of the opened can.

Figure 2—shows the reverse surface (or oil's eye view) of the top of the can.

Figure 3—shows the enlargement of Figure 2.

Figure 4—shows a part of the edge of the jagged sheared opening, enlarged 75 times. Here you may see several metal particles 10 and 20 microns in size ready to break off and contaminate the oil.

One might argue that these particles are still attached to the parent metal, but it is probably quite safe to assume that many particles of this size or larger have been produced and rubbed off by the cutting action of the cutter-funnel and during its removal. These particles would not of course be seen, but they would enter the oil system during this or the next filling.

Figure 5—To illustrate this, Figure 5 shows an enlargement of a sliver of metal 0.10" long which was attached to the parent metal very lightly and fell off during the preliminary visual examination.

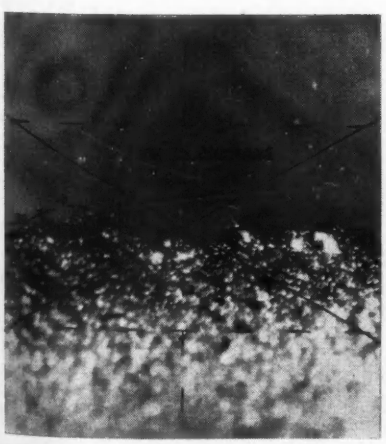


Figure 4

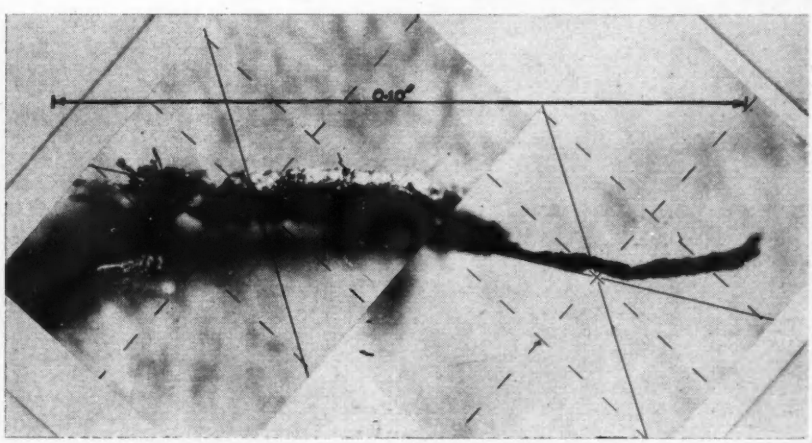


Figure 5

In automobile crankcases, one may question whether the smaller particles will do any harm; but if the same method of opening tins is to be used for aircraft hydraulic fluid containers, it seems pointless to insist on absolute purity for the oil and then to introduce swarf into the fluid during the filling operation.

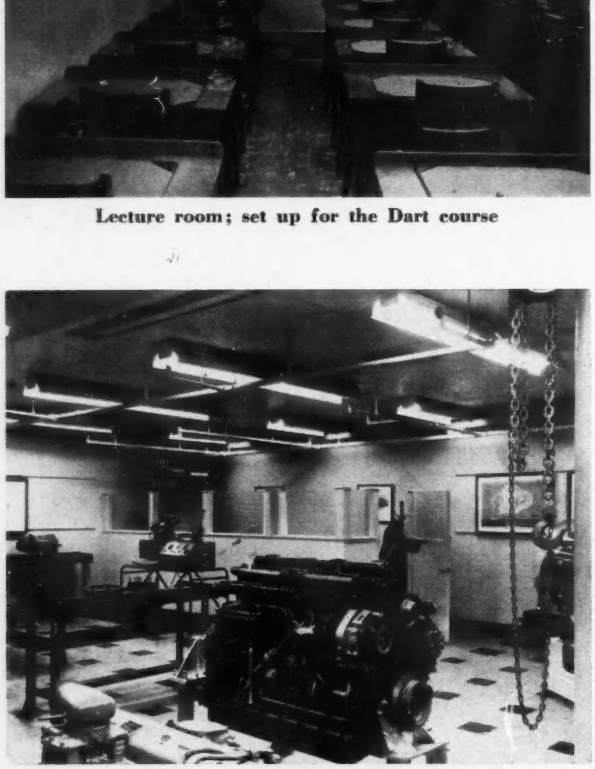
It would appear that filtration of the oil between the filler cap of the tank and hydraulic system components is essential if one is to continue to require the cleanliness of oil now specified.

Ottawa — M. S. KUHRING

A Course on the Dart Engine

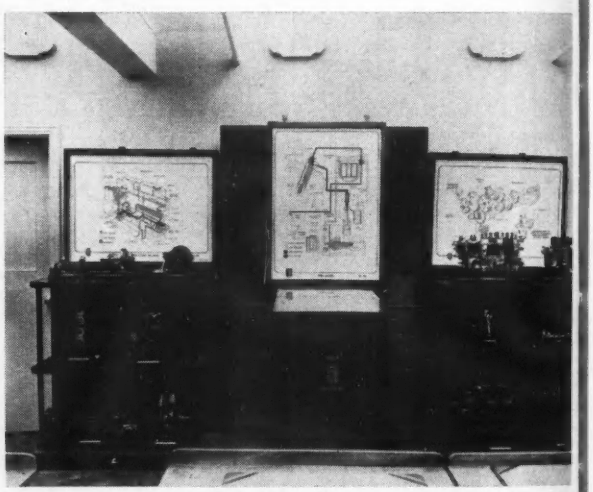
ON the 6th January, 1958, four members of the C.A.I. reported for a one week Dart course at the Rolls-Royce Training School.

Lecture room; set up for the Dart course



Demonstration room: the Rolls-Royce Diesel is in the foreground

Attending the Dart course (l to r) M. D. Mugford, Genaire Ltd., A. W. Stewart, Carriere and MacFeeters Ltd., C. Kapica, Fenco Montreal, W/C P. de L. Markham, RCAF



Standard training pack-up; in this case, equipped for the Rolls-Royce Diesel. Provision is made for training charts and demonstration models and a projection screen can be mounted over the blackboard in the centre

Course No. 32, as it was known officially, was welcomed by Mr. K. H. Parker, who is in charge of Training, and was then handed over to Mr. J. Morton for the actual instruction. During the course, approximately 200 pages of diagrams and notes were issued and the subject matter was covered by Mr. Morton in four days, as opposed to nine days which is the time taken on the normal two-week course. That the instruction was very good indeed goes without saying and those attending were all impressed with Mr. Morton's ability to answer all questions – even those which were not directly associated with the subject!

It is a pity that the response to this course should have been so small but it is hoped that the excellent report of those who did attend will stimulate interest in any future venture of this nature by the C.A.I.

Ottawa — W/C P. DE L. MARKHAM

C.A.I. LOG

SECRETARY'S LETTER

THANK YOU

LATE though it is – and I blame the ponderous workings of the Journal for the delay – I should like to thank all those who sent us Christmas greetings. The place was very gay with cards and I assure you that the good wishes which they expressed were greatly appreciated by all of us at Headquarters.

STAFF

On the subject of Headquarters staff, there have been so many changes during recent months that it is about time I made a few introductions. Members are liable to receive letters or phone calls from any one of us and it is always helpful in such circumstances to know who's who.

Mrs. G. G. Ross is Assistant to the Secretary. She has been with us since the very beginning and the C.A.I. could scarcely run without her. Apart from being my right hand in almost everything I do, she is primarily responsible for the Journal.

W/C H. L. Taylor is Assistant Secretary. He has not been with us very long but he is already relieving me in many ways. I hope that all our members will soon get to know him well, because he will take over a great deal of the membership work.

The three other members of the staff are Miss J. M. White, who is my secretary, Miss R. A. Cosgrove, who is W/C Taylor's, and Miss M. L. Grant, who does an admirable job of keeping our many records in good shape.

I hope that this team will now settle down, with no more changes for a while, and that we shall be able to develop and expand the work that we are trying to do for the Institute.

OUR PROGENITOR SOCIETIES

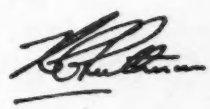
Now that the Secretary of State has approved the recent amendment to the Bylaws (thereby bestowing official confirmation), as reported on page 47 of this issue, I can reply to a question which was raised once or twice during the balloting – "why does the specially reduced rate of dues no longer apply to members who are also members of other societies contributing to our support?" The answer is that there are no longer any such societies.

When the Institute was formed the Royal Aeronautical Society, the Institute of the Aeronautical Sciences and the Engineering Institute of Canada very generously agreed to support us financially for a few years, to give us an opportunity to find our feet and to develop our own resources. It was never intended that their annual grants should be permanent – except in their effect; and their effect has certainly been that the C.A.I. has become established as it never would have been without them.

We have now told these three societies that we feel strong enough to support ourselves financially. I am sure that all our members will agree with the Council's decision that the Institute should fly by itself, now that it is tolerably fledged.

But this financial independence does not mean that we feel that we can do without the advice and guidance which the three older societies have always been so willing to give us whenever we asked for it. We are not nearly ready to forego that sort of help and I know that we can count on it in the future.

Probably very few members of the Institute realize how much, in hard cash and in advice and encouragement, we owe to the R.Ae.S., the I.A.S. and the E.I.C. You must just take my word for it and be properly grateful.



MID-SEASON MEETING

Programme

February 27th Morning 9.00 a.m.

TEST FLYING

Chairman
D. H. ROGERS
Chief of Flight Operations
Avro Aircraft Limited

Flight Testing in the Space Age
A. W. BLACKBURN
Engineering Test Pilot, Los Angeles Division
North American Aviation Inc.

Some Problems in Airline Turbine Transport Operation
CAPT. R. J. BAKER
Flight Test Engineer-Pilot, Flight Test Engineering
Trans-Canada Air Lines

Supersonic Aircraft Escape Systems
R. H. FROST
Vice-President
Stanley Aviation Corporation

Afternoon 2.00 p.m.

OPERATIONS AND SAFETY

Chairman
E. L. BUNNELL
Chief Test Pilot
Bristol Aircraft (Western) Limited

Search and Rescue in Canada
S/L J. W. BORDEN
Officer Commanding
121 Communications and Rescue Flight

F/L W. G. DAVISON
Air Force Headquarters
Royal Canadian Air Force

Airline Planning for Jets
J. A. STERN
Staff Engineer
United Air Lines Inc.

Future Developments in Air Traffic Control
I. G. BARROWMAN
Superintendent, Air Traffic Control Procedures

B. CAPEL
Specialist, Air Traffic Control Future Requirements
Department of Transport

Evening 8.00 p.m.

SPECIAL SESSION

Chairman
H. H. OLLIS
General Manager
Aviation Electric Pacific Limited

Selection of the Vickers Vanguard by Trans-Canada Air Lines
J. T. DYMENT
Chief Engineer
Trans-Canada Air Lines

February 28th Morning 9.00 a.m.

ELECTRICS AND ELECTRONICS

Chairman
S/L A. E. KELLY
Senior Telecommunications Officer
R.C.A.F. Station Cold Lake

The Doppler Sensor in Air Navigation
K. C. M. GLEGG
Special Assignments Engineer
Canadian Marconi Company

The Decca Navigator System
L. J. DENNETT
General Manager

R. C. MARTIN
Systems Engineer
Decca Navigator Company Limited

Electronics and T.C.A.
C. J. CAMPBELL
Director of Telecommunications
Trans-Canada Air Lines

Afternoon 2.00 p.m.

INSPECTION AND MAINTENANCE

Chairman
J. A. CLARKE
Manager, Quality Control
Northwest Industries Limited

Aluminum Corrosion—An Unnecessary Evil
J. R. BEALE
Senior Engineer, Sales Development Division
Aluminum Company of Canada Limited

Non-Destructive Testing of Aircraft Materials
H. W. GRANT
Manager, Engineering Services
Standard Aero Engine Limited

Airline Planned Maintenance
R. J. MCWILLIAMS
Superintendent of Planning, Maintenance and Engineering Division
Canadian Pacific Air Lines Limited

Evening 7.00 p.m.

DINNER

Chairman
G/C H. R. FOOTTIT
President, Canadian Aeronautical Institute
Director of Aircraft Engineering, R.C.A.F.

Principal Speaker
E. C. WELLS
President, Institute of the Aeronautical Sciences
Vice-President, Engineering, Boeing Airplane Company

MEETINGS

MID-SEASON MEETING

Notices giving the full programme of the Mid-season Meeting have been sent to all members of the Institute; the programme is also set out on the opposite page.

ABSTRACTS

The following are abstracts of some of the papers to be presented at the Mid-season Meeting.

Flight Testing in the Space Age

A. W. Blackburn—
North American Aviation

In the belief that the highly touted missile arsenal must give way to more truly versatile manned weapons systems just as certainly as the beeping sputniks must be followed by men in space, the gathering of data through flight test for hastening these developments is investigated. The problems presented by a world-wide test arena, stability and control in space, re-entry, temperature effects and human frailties are discussed if not resolved. In any event, the manned concept's the thing.

Some Problems in Airline Turbine Transport Operation

R. J. Baker—T.C.A.

The author claims that there is nothing very new in this paper on some of the operational problems which are being encountered now and may be expected in airline turbine transport operation. It is only an attempt to gather together some of the more pertinent problems which he feels are associated with these new turbine-engined aircraft and discuss them in the light of our present knowledge. Experience only will indicate whether or not the present solutions are correct. The length of time will not permit a study in any great detail but the paper presents a brief review of some of the problems involved in handling the aircraft from ramp to ramp around the world.

Supersonic Aircraft Escape Systems

R. H. Frost—Stanley Aviation

Circumstances associated with the widely publicized supersonic ejection of a North American test pilot in 1955 gave tremendous impetus in the United States to development of escape systems able to cope with such emergencies. This paper describes the significant advances that have resulted therefrom, with special emphasis on the supersonic upward and downward ejection seats being developed by a joint Air Force-Industry Committee. In addition, mention is made of other approaches to solution of the problem, such as the rocket catapult and encapsulated seat. Capsule development and predictions of space vehicle escape systems complete the presentation.

Search and Rescue in Canada

S/L J. W. Borden and
F/L W. G. Davison—R.C.A.F.

The subject is introduced by giving something of the historical background that led to the establishment of the Search and Rescue service. The SAR task is then defined and requirements for adequate SAR coverage are discussed. An outline of the Search and Rescue organization follows, with particular reference to the methods used to provide for the requirements discussed earlier.

A description of the various SAR aircraft and related equipment is then given. Finally, comment is made on recent developments in this field, including a discussion of equipment that may be used in the future.

Airline Planning for Jets

J. A. Stern—United Air Lines

The airlines' planning for jet operations runs the gamut from financing to food handling. This paper outlines the planning in the numerous technical areas which is necessary before efficient and economical commercial jet operations can be commenced. Specifically discussed are the airport facilities including the runway, ramp and terminal requirements. The optimum operating procedures, both ground and flight, are considered in the light of available data and experience. Maintenance planning is dealt with as it is essential for the efficient utilization and scheduling of large aircraft. While this paper chiefly deals with the technical aspects of the planning necessary for the introduction of commercial jet transports, reference is made to non-technical areas, such as passenger service requirements, schedule planning and sales promotion, where these have a bearing upon planning in strictly technical areas.

Future Development in Air Traffic Control

I. G. Barrowman and B. Capel—D.O.T.

The paper provides a general indication as to what may be expected of ATC in the immediate future and includes a traffic control forecast for the next ten years. The surveillance radar programme is outlined; the problems of integrating jet with other aircraft and the plans for handling jet transport operations also are summarized. The paper touches upon automation, including computation, regulations and regimentations, the possibility of employing new techniques, such as visual displays, data transfer systems; point source versus area coverage navigational systems and the expected roles of self-contained navigation systems, proximity warning systems and secondary radar.

The Doppler Sensor in Air Navigation

K. C. M. Glegg—Can. Marconi

Since early in 1956, the security restrictions covering Doppler drift and groundspeed measuring airborne radar (Doppler Sensor) have been gradually removed. As a result, various groups, in addition to the military, have begun considering potential uses of the device which promises to put dead reckoning navigation on an essentially new footing with regard to accuracy.

The object of this paper is to present a general description of the device, with particular attention to the various types of errors which find their way into the measurement of drift and groundspeed. The results are then combined with other data relating to heading references to produce accuracy numbers which should allow the professional navigator to assess the utility of the device.

Electronics and Trans-Canada Air Lines

C. J. Campbell—T.C.A.

The role of electronics in the operations of civil aviation in general, and Trans-Canada Air Lines in particular, is constantly increasing in importance. Air/ground communications, point-to-point communications and electronic navigational systems comprise the "nerve system" of airborne operations. New techniques have been developed and systems evolved during the past ten years which are of interest and will be reviewed. The trend is towards automation in these fields and we take a brief look at where we may be going in this respect.

The application of electronics in the airline is not limited, however, to airborne operations but is rapidly being adapted to other phases of the airline business. The wide-spread interest in

and progress with automatic reservations and electronic data processing will also be covered.

Aluminum Corrosion–An Unnecessary Evil

J. R. Beale–Aluminum Co.

Aluminum alloys have been used successfully in the aircraft industry for many years. This success has been due to proper design considerations which result in keeping the conditions conducive to corrosion at a minimum.

This paper outlines the basic causes and types of corrosion which might be encountered, together with a discussion of methods of corrosion prevention. This includes the various levels, from raw material to aircraft in use. Under proper usage conditions aluminum alloy parts should have a continuing life free from the costly toll of corrosion.

Non Destructive Testing of Aircraft Materials

H. W. Grant–Standard Aero Engine

The constant research, testing and production of materials with higher stress levels and high factors of safety by the aircraft industry over the past 30 years has been a predominant factor in the development and perfection of non destructive testing.

There are numerous highly effective methods in present day use which include magnetic particle inspection, fluorescent penetrant inspection, radiography ultrasonics and many others. Each has been developed to fulfill a need either with respect to material or application.

The magnetic particle and fluorescent penetrant methods of testing are two which are used extensively both in the manufacturing and maintenance of aircraft and engine components.

The paper concerns the techniques and application of these methods.

Airline Planned Maintenance

R. J. McWilliams–C.P.A.L.

The attempt is made to show how one company has evolved a system for controlling its flying and maintenance operations to achieve greater economy of its resources, and to show how this economy may be further improved by more attention to those details which have an adverse effect on maintenance costs.

The paper deals with the subject in three steps; outlining the objectives; describing the systems and procedures developed to meet the objectives; and closing with a discussion of the major problem areas including some suggestions for the improvement of planning functions.

SUSTAINING MEMBERS

NEWS

The Fairey Aviation Company of Canada Limited has developed a Safety Ohmmeter for wire bridge types of missile fuzes. This portable instrument measures resistances in four ranges up to 10K ohms. It is powered by a photo cell of unlimited life to ensure that short circuit current will not exceed 10mA and open circuit voltage 0.7v. Fundamental tolerances are 0.3%. Measurement sensitivity is determined by intensity of light incident on the photo cell. Corrected potentiometer in Wheatstone bridge circuit and scale drum of 6.5 in effective scale length divided into 100 divisions ensures high scale reading accuracy. Meter movement sensitivity is 2.4 μA/0.1" at centre. This instrument can also be used in other fields of electrical measurement.

In addition, the company has developed a Microfilter specifically designed to give "absolute" first pass removal from fluids of all foreign particles above 4 microns in size. The sintered bronze elements are constructed of uniform pellets and the microfilter is designed to cover system pressure ranges from 60 psi to 5,000 psi with fluid flows up to 30 gpm at 5,000 psi. The filter element can withstand a pressure differential of 450 psi and designs are available for operating in fluids over a wide range of temperatures. The filter can be supplied with bypass valves and automatic warning features to indicate clogged elements. Now in quantity service in aircraft and other fields, these units can be designed to meet individual operating conditions.

Aviation Electric Limited have announced the completion of an addition of 35,000 sq ft to their Montreal plant. The new area houses an air-conditioned manufacturing machine shop and special test facilities for jet fuel controls, fuel flowmeters and hydraulic and other equipment associated with high inflammable liquids. The entire new section, which has been added to the existing plant in the form of a wing, has been equipped with air filtering, humidity and temperature control. Temperature will be maintained at a standard 72°F, with tolerance of ±1°. This will permit the manufacturing of precision instruments, accessories and components for aircraft and other industries requiring extremely close tolerances. The addition of this new wing brings the total floor area of the Montreal plant to 150,000 sq ft.

Simmonds Aerocessories of Canada Ltd. advise that they have moved their offices from Montreal to Hamilton, where their new address is

637 Parkdale Ave. N.,
Hamilton, Ont.

Computing Devices of Canada Ltd. draw attention to the announcement that an air and sea navigation system, already in use over a million square miles in eastern Canada, will be evaluated in the U.S.A. by the Airways Modernization Board.

Installation of a Bendix-Decca Navigator chain in the New York area follows installation during the past year of four chains covering the area from east of Newfoundland to Montreal. Though the Canadian chains are being evaluated for both marine and air applications, emphasis on the New York chain will be for helicopter use.

Computing Devices, the firm responsible for Bendix-Decca in Canada, predict that helicopter operations will ultimately become an important use of the system in Canada, though current interest in Canada is directed more at conventional air and marine uses, and such specialized applications as forest spraying.

The objective of the New York test is to determine to what extent a hyperbolic system will expedite helicopter operations in a high density terminal area. Bendix-Decca was selected because it is the only such system now in a state of operational readiness.

MEMBERS

NEWS

A/V/M A. Ferrier, Hon. F.C.A.I., has retired from his position of Assistant Secretary General, Air Navigation, International Civil Aviation Organization.

P. E. Halsey, M.C.A.I., Assistant to the Director of Engineering, De Havilland Aircraft of Canada Ltd., has been appointed to the Board investigating the crash of a Maritime Central Airways Ltd. DC-4 aircraft in August, 1957, near Issoudun, P.Q.

A/V/M F. G. Wait, M.C.A.I., has left Fleet Manufacturing Ltd. to take up an appointment as Assistant to the Vice-President of Enheat Aircraft, the Aircraft Division of Enamel & Heating Products Ltd.

A. J. Comeau, Technical Member, formerly with Avro Aircraft Ltd., has taken up the position of Assistant to the Electronics Engineer at the Garrett Manufacturing Co. of Canada.

ADMISSIONS

At a meeting of the Admissions Committee, held on the 2nd January, 1958, the following were admitted to the grades shown.

Associate Fellow

W/C A. Chornobrywy (on regrading from Member)

D. G. Cramb, Chief Mech. Development Engineer, Orenda Engines Ltd., Box 4015, Terminal A, Toronto, Ont.

F. D. M. Williams, Chief Controls Develop. Engineer, Orenda Engines Ltd., Box 4015, Terminal A, Toronto, Ont.

Member

Dr. D. C. Baxter, Assistant Research Officer, Mech. Engr. Div., National Research Council, Ottawa, Ont.: *45 Belgrave Rd., Ottawa, Ont.*

J. A. Clarke, Manager, Quality Control, Northwest Industries Ltd., Edmonton, Alta.: *9739-146th St., Edmonton, Alta.*

Sgt. A. J. Ducharme, RCAF, Supervisor Aeronautical Drafting, AMC/SOEPES, Ottawa, Ont.: *39 Via Venus, Ottawa 2, Ont.*

F/O J. J. Francis, RCAF, Range Photo Officer, Primrose Lake, Alta.: *Box 1257, MPO 503, Grande Centre, Alta.*

D. J. Freshwater, Design Checker, Orenda Engines Ltd., Malton, Ont.: *31 Tichester Rd., Apt. 402, Toronto 10, Ont.*

S. H. E. Garside, Assistant to the General Manager, York Gears Ltd., 825 Caledonia Rd., Toronto 10, Ont.

K. Lancaster, Assistant to Vice-President Engineering, Renfrew Aircraft & Engineering Co., Ltd., Renfrew, Ont.

R. W. Moore, Design Engineer, Avro Aircraft Ltd., Malton, Ont.: *3126 Princess Blvd., Burlington, Ont.*

F/O M. H. Niblett, RCAF, Technical/AE Officer, RCAF Stn. Trenton, Ont.: *66 Crown St., Trenton, Ont.*

L. W. Smith, Assistant General Manager, Bristol Aero Engines Ltd., Montreal North, P.Q.: *38 Lilac Ave. South, Dorval, Montreal, P.Q.*

B. C. Stonehill, Aerodynamicist, Dilworth Ewbank, Toronto, Ont.: *37 Harlow Cres., Thistletown, Ont.*

Technical Member

J. H. Ford, Sales & Service Representative, Aviation Electric Ltd., Montreal, P.Q.: *334 North Cote Rd., Rosemere, P.Q.*

N. C. Goudriaan, Drawing Checker, Engr. Dept., Northwest Industries Ltd., Edmonton, Alta.: *11928 - 141st St., Edmonton, Alta.*

P. E. M. Hynes, Electrical Draftsman, Design Office, The Fairey Aviation Co. of Canada Ltd., Dartmouth, N.S.: *52 Tulip St., Dartmouth, N.S.*

F/O A. A. MacDonald, RCAF, Tech. Control Officer, Maint HQ, 3(F) Wing, CAPO 5055, Canadian Armed Forces Europe.

A. J. D. MacKinnon, Electrical Designer, The Fairey Aviation Co. of Canada Ltd., Dartmouth, N.S.: *518 Robie St., Halifax, N.S.*

J. P. Mentink, Designer, Jarry Hydraulics, Montreal, P.Q.: *3946 Ross St., Verdun, P.Q.*

M. A. Rowe, Foreman of Radio and Electrical Maint., Canadian Pacific Air Lines Ltd., Vancouver Airport, B.C.: *1184 West 64th Ave., Vancouver 14, B.C.*

R. C. Walker, Inspector, Assembly & Test, Lucas-Rotax Ltd., Scarborough, Ont.: *30 Arbutus Cres., Scarborough, Ont.*

A. W. Wright, Technical Assistant, Dowty Equipment of Canada Ltd., Ajax, Ont.: *19 Burcher St., Ajax, Ont.*

Student

H. D. Schiller, University of Toronto, Toronto, Ont.: *5 Lark St., Toronto 8, Ont.*

* * *

At a meeting of the Admissions Committee, held on the 16th January, 1958, the following were admitted to the grades shown.

Associate Fellow

E. B. Moss, Technical Director, Canadian Applied Research Ltd., 1500 O'Connor Dr., Toronto 16, Ont.

Member

D. J. Carey, Aerodynamics Engineer, Canadair Ltd., Montreal, P.Q.: *3465 Ridgewood Ave., Montreal 26, P.Q.*

WO II M. W. Corbett, RCAF, Aircraft Maint. Superintendent, AMCHQ/-RCAF Stn. Rockcliffe, Ont.: *48 Fentiman Ave., Ottawa 1, Ont.*

R. C. Drinkwater (on transfer from Technical Member)

P. Harper, Engineer A, Canadair Ltd., Montreal, P.Q.: *280 Constantin St., Ville St. Eustache, P.Q.*

D. N. Harkness (on transfer from Technical Member)

E. E. Jacobs, Aviation Engineer, The Texas Company, 1730 Clifton Place, Minneapolis 3, Minn.

S/L A. E. Kelly, RCAF, Stn. Telecommunications Officer, RCAF Stn. Cold Lake, Alta.: *MPO 503, Grande Centre, Alta.*

LCDR(E) J. D. Newton, RCN, Deputy Director of Air Engineering, RCN Headquarters, Ottawa, Ont.

R. Parker, Liaison Engineer, Honeywell Controls Ltd., Aeronautical Div., Leaside, Ont.: *274 Melrose Ave., Toronto 12, Ont.*

A. T. Smith, Jr. Stress Engineer, Avro Aircraft Ltd., Malton, Ont.: *101 Orchard View Blvd., Toronto 12, Ont.*

R. J. Thatcher, Inspection Superintendent, Trans-Canada Air Lines, Winnipeg, Man.: *35 Baffin Cresc., St. James, Man.*

R. L. Wardlaw, Research Officer, National Research Council, Ottawa, Ont.: *Box 556, R.R. 1, Ottawa, Ont.*

G. A. Worden, Powerplant Engineer, Canadian Pacific Air Lines Ltd., Vancouver Airport, B.C.: *4030 Carson St., South Burnaby, B.C.*

Technical Member

W. J. Bice, Assembler, Lucas-Rotax Ltd., Toronto, Ont.: *32 Fraserton Cresc., Scarborough, Ont.*

F/O L. F. Bolger, RCAF, Staff Pilot, CEPE/AAED, RCAF Stn. Cold Lake, Alta.: *c/o Officers Mess, MPO 503, Grande Centre, Alta.*

***Technical Member* (cont.)**

Lt.(P) M. H. Brayman, RCN, Project pilot VX10, RCN Air Station, Shearwater, N.S.

R. G. Day, Commercial Representative, Northwest Industries Ltd., Edmonton, Alta.: *13024 - 122 Ave., Edmonton, Alta.*

Lt.(E)(AE) T. A. P. Eyre, RCN, OC Aeronautical Engineering Div., RCN Air Station, Shearwater, N.S.

R. R. J. Fleming, Technical Officer, National Research Council, Flight Research Br., Uplands, Ont.

F. X. Gregor, Tech. Assistant, Hydraulic Dev. Dept., Lucas-Rotax Ltd., Toronto, Ont.: *34 Parkview Hill Cresc., Toronto 16, Ont.*

T. Gunzler, Demonstrator, University of Toronto, Toronto, Ont.: *12 Eastmount Ave., Toronto, Ont.*

D. F. Harrison, Draftsman, Canadian Pacific Air Lines (Repairs) Ltd., Lincoln Park, Alta.: *729 - 25 Ave. N.W., Calgary, Alta.*

Lt.(E)(AE) W. B. Hotsenpiller, RCN, OC Quality Control Div., RCN Air Station, Shearwater, N.S.: *Wardroom, R.C.N.A.S. Shearwater, N.S.*

S. T. Le Bedis, Test Pilot, Engineering Flight Test Dept., Canadair Ltd., P.O. Box 6087, Montreal, P.Q.

D. J. Marsden, Jr. Research Officer, National Research Council, Ottawa, Ont.: *235 Mart Circle, Manor Park, Ottawa, Ont.*

Technician

W. H. Davison, Draftsman, De Havilland Aircraft of Canada Ltd., Toronto, Ont.: *19 Olympus Ave., Swansea Village, Toronto 3, Ont.*

W. A. Harper, Tech. Assistant, The Fairey Aviation Co. of Canada Ltd., Dartmouth, N.S.: *35 Pleasant St., Dartmouth, N.S.*

M. E. Smets, Assembler, De Havilland Aircraft of Canada Ltd., Toronto, Ont.: *105 Downsview Ave., Downsview, Ont.*

Student

F/C G. P. Stevens, RCAF, University of Manitoba, Winnipeg, Man.: *618 Oak St., Winnipeg 9, Man.*

APPOINTMENT NOTICES

The facilities of the Journal are offered free of charge to individual members of the Institute seeking new positions and to Sustaining Member companies wishing to give notice of positions vacant. Notices will be published for two consecutive months and will thereafter be discontinued, unless their reinstatement is specifically requested. A Box No., to which enquiries may be addressed (c/o The Secretary), will be assigned to each notice submitted by an individual.

The Institute reserves the right to decline any notice considered unsuitable for this service or temporarily to withhold publication if circumstances so demand.

Positions Vacant

Electrical Draughtsmen: Vacancies for several men with at least 3 years experience of draughting on electrical and electronic installations in aircraft. Applicants should have technical education to British Ordinary or Higher National Certificate (Electrical) level and knowledge of Canadian and U.S. aircraft electrical standards. Company offers medical and hospitalization benefits and, after a qualifying period, a Pension Plan. Write giving age, qualifications, full details of experience and salary required to Industrial Relations Manager, Northwest Industries Limited, Box 517, Municipal Airport, Edmonton, Alta.

Electrical Engineer: Graduate or equivalent with a minimum of 5 years' experience in electrical and electronics equipment in aircraft field. Autopilot experience an asset. Required to control design and prototype development in instrumentation and test equipment field and to advise on production queries. Apply to Personnel Manager, Aviation Electric Limited, 200 Laurentian Blvd., Montreal, P.Q.

Hydraulics Engineer: Graduate or equivalent with a minimum of 5 years' experience in aircraft or missile hydraulic field. Servo valve experience an asset. Required to control design and prototype development and to advise on production queries. Apply to Personnel Manager, Aviation Electric Limited, 200 Laurentian Blvd., Montreal, P.Q.

Sales Representative: Bilingual with 5 to 10 years' practical aircraft maintenance experience, particularly on bush type aircraft. Must be familiar with aircraft accessories and instruments. Extensive travelling is required. Apply to Personnel Manager, Aviation Electric Limited, 200 Laurentian Blvd., Montreal, P.Q.

Technical Sales Representatives: To work with airframe and engine design engineers in applicating fluid pumps and valves. Candidates should have a good understanding of fuel, hydraulic, pneumatic, oxygen, anti-icing and airconditioning systems. Engineering degree desirable but not essential. Vacancies exist in both Montreal and Toronto areas. Applications should be submitted in the first place by letter to the Aviation Manager, Railway & Power Engineering Corporation, Ltd., P.O. Box 880, Montreal, Que.